

Heat Transfer in Aerospace Propulsion

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SUMMARY

This paper presents an overview of heat transfer related research in support of aerospace propulsion, particularly as seen from the perspective of the NASA Lewis Research Center. For this paper, aerospace propulsion is defined to cover the full spectrum from conventional aircraft power plants through the Aerospace Plane to space propulsion. The conventional subsonic/supersonic aircraft arena, whether commercial or military, relies on the turbine engine. A key characteristic of turbine engines is that they involve fundamentally unsteady flows which must be properly treated. Space propulsion is characterized by very demanding performance requirements which frequently push systems to their limits and demand very tailored designs. The hypersonic/transatmospheric flight propulsion systems are subject to very severe heat loads and the engine and airframe are truly one entity. The impact of the special demands of each of these aerospace propulsion systems on heat transfer will be explored in this paper.

INTRODUCTION

Aerospace propulsion presents a demanding challenge to the heat transfer engineer. The designer seeks to transform the chemical energy of the fuel into the useful work of propulsive thrust at maximum effectiveness. To do this the propulsion system must be operated at very high temperatures and pressures with very few parasitic losses, all concentrated into as small a package as possible to minimize weight. These requirements are frequently in conflict and the conflict often centers around the ability to protect the propulsion surfaces from this hostile thermal environment.

One approach, being aggressively pursued today on a national scale, is to develop materials capable of withstanding these hostile environments and offer an essentially adiabatic surface that will not melt or lose its structural integrity. Failing this, the other approach is to actively cool the exposed surfaces. Both of these require the heat transfer engineer, although the cooled system is probably the more demanding and is a major activity in today's propulsion system designs.

There are numerous figures of merit for the performance of an aerospace propulsion system, depending on the mission of the flight system: specific fuel consumption (SFC); thrust-to-weight ratio (T/W); and specific impulse (Isp), to name a few. Specific impulse, which is the ratio of the thrust to the weight (not mass) flow rate of the propellants, is particularly useful in deciding the engine requirements for a mission since it relates the engine and propellants to the total system to be propelled. An example of a hypothetical Earth-to-orbit mission, taken from charts prepared by the Air Force Wright Aeronautical Labs (AFWAL) is shown in figure 1, illustrating the range of Isp needed to do

the mission. A companion figure 2 shows the engine selections available for such a mission using either hydrogen or hydrocarbon fuels. Another interesting look at engine selection is shown in figure 3. Notice that one of the boundaries is heat transfer; a limit designers are constantly striving to move, in order to expand an engine's operating corridor.

Charts such as these show that aerospace propulsion has very complex requirements, which demand a wide variety of very high performance engines. Thus, for purposes of this paper aerospace propulsion is defined in its broadest context, ranging from the propulsion of subsonic and supersonic aircraft through propelling vehicles to the deep regions of space.

The paper will begin by examining some interesting aspects of conventional subsonic/supersonic aircraft propulsion. It will move then to the other end of the spectrum to space propulsion. The final section will bridge the gap from aircraft to space in true "aerospace" propulsion - transatmospheric flight propulsion. Such a vast territory by necessity requires some narrowing. Thus, it is the authors' intention to select a few limited, but hopefully interesting, insights to the type of heat transfer problems facing designers today in aerospace propulsion. This paper is not intended to be an in-depth state-of-art review on any one subject. In order to facilitate assembling the material the authors have drawn on the work of NASA and NASA contractors, with supplements from the general literature where necessary to make a point. In doing this, the authors wish to acknowledge in advance the very sizable research activity in aerospace propulsion heat transfer, even though only a small part of it is referenced.

SUBSONIC/SUPERSONIC PROPULSION

The propulsion system for subsonic and low end (Mach 2 to 4) supersonic flight is the turbine engine, now entering its second half century. In both the civil and military arenas the engine designers have set their sights on achieving a major advancement in turbine engine technology by the turn of the century. An example in the commercial arena, focusing on very high propulsion efficiency, is an ultra-high bypass ratio engine shown in figure 4, basically a shrouded turboprop. In the military arena the focus is on very high performance and very lightweight, as for example shown in figure 5. A common feature, which is of particular interest to heat transfer engineers, is that the core of the engine - the hot section - will be operated at very high temperatures, approaching stoichiometric.

The hot section includes: the combustor, the turbine, and the exhaust nozzles, and also some allied components. They all offer special heat transfer challenges. However, the turbine is especially demanding, so we will concentrate on it. The technical challenges are many. They include: the proper treatment of time dependency; turbulence (closure) modeling; film cooling; complex internal passages; rotational effects; and also new materials and thermal stress analysis. We will explore the first topic in some depth, albeit limited, and touch on the others.

A central fact to be addressed in turbines is that the flow in them is fundamentally unsteady. Most fluid mechanics and heat transfer analyses directed at turbines today are steady flow analyses. This is okay as long as

we understand that "steady" really means time average and that the averaging must be properly done.

Before pursuing this further, let us examine some of this unsteady flow. Laser anemometry measurements by Suder et al. (1987) and Hathaway et al. (1987) in the wake of a compressor rotor are shown in figure 6 and display the "turbulent" kinetic energy distribution in the downstream stator passage. The unsteadiness in this flow has several components. One is the change in mean velocity due to the passing airfoils. This has been removed from figure 6 by ensemble averaging. The remaining disturbances are labelled "unresolved unsteadiness" by Hathaway since they contain both classical stochastic turbulence convecting through the machine and vortex shedding from the passing blades. These cannot be removed from this data, because they occur at a frequency different than the blade passing frequency. By a different analysis of similar data Hathaway et al. (1986) established that turbomachinery airfoils have shed vortices despite their very high Reynolds numbers and near sonic velocities. Another view of turbomachinery unsteadiness has been provided by Sharma (1983) in the wake of a turbine rotor. A computational view of flow unsteadiness is provided by Jorgenson and Chima (1988), and shown in figure 7 for the Space Shuttle Main Engine (SSME) fuel turbine. The variable dark areas near the front of the rotor blades represent unsteady velocities or pressure forces on these blades as they pass through the upstream stator wakes.

What does all this mean for heat transfer? The question can probably be best answered by looking at some remarkable work being performed by Dunn (1982, 1988). Dunn's facility is a shock tunnel in which a full turbine stage can be operated at fully scaled conditions for a period of 30 ms. Figure 8 shows a turbine instrumented with very high response (100 kHz) thin film heat flux gages and the corresponding heat flux data on the suction side of the blade for one stator vane passing event, which lasted for 97 μ s. Notice that the heat flux jumps from laminar to turbulent levels during every cycle. Data like these have convinced researchers that they will eventually need full flow field and heat transfer data taken in the same facility. This is almost impossible in the shock tunnel. Blowdown rigs such as described by Guernette (1988) offer a possibility. Another approach by O'Brien (1986, 1988a, 1988b) is to employ a simulated turbine rotor wake rig. Data from these experiments are shown in figure 9. By combining the flow field and heat flux information and employing the widely used Lowery and Vachon (1975) correlation for the effect of free-stream turbulence on heat transfer to the stagnation region of a circular cylinder, one finds surprisingly good agreement for the effect of the rotor wake on the downstream stator stagnation heat transfer. It should be noted that this result is an average result, but that the average is an ensemble based on the characteristics of the flow and the blade passing unsteadiness has been retained.

Other averages are even more global, yet they are responding to the variety of unsteadiness in the flow. An example is heat transfer data by Dring et al. (1986) and Blair et al. (1988) acquired in a large low speed turbine. The data, shown in figure 10, indicate a very substantial effect of inlet turbulence on the first vane row. The rotor, subjected to the stator wakes, shows very high transfer, but the residual effects of the inlet turbulence seem to be much less dramatic. Data in the second vane row, downstream of the first stage, not shown herein, exhibited a high degree three dimensionality. This may be consistent with the vortical flows seen by Sharma (1983) in the same

machine. Measurements of the time-averaged total pressure distribution exiting the first stage rotor of this same turbine have been made by Joslyn and Dring (1988) and are shown in figure 11(a). A very complex flow passage structure can be seen. Working closely with these data Man Mohan Rai at NASA Ames has attempted to calculate these flows with a time accurate three-dimensional viscous rotor-stator interaction code, Rai (1987). The results of Rai's calculations, shown in figure 11(b), are quite similar to the data. Work is continuing to refine the grid and blade/vane count, turbulence modeling, averaging techniques, and treatment of the inlet conditions.

As an aside, one should note that much of what is seen in the above experimental and analytical figures would not be possible without very sophisticated computer graphics. The current ability to measure and compute is so great that it is rapidly outstripping our ability to absorb and comprehend the information. Graphic's and animation, as seen above, allow insights into the data never seen before. It is an area requiring attention. At the same time it is an area requiring care. The nicer it looks the more believable it is. We must never forget that the result is only good if it properly represents the physics. Nevertheless, it is important to note that computer graphics have moved beyond nice presentation into a valid tool for aiding the understanding of the physics.

All of this suggests a strong need for a rational approach for relating the time resolved information to the time average. One such methodology is provided by Adamczyk (1985). Adamczyk performed a sequence of averaging processes on the Navier-Stokes equations, based on the unsteady physics in a turbomachine. He calls this the average passage method. Three successive averages for random unsteadiness, periodic unsteadiness, and unequal blade count from row to row yields a set of equations which require considerably less computer power to solve, but also introduce many new unknowns which must be determined in some other way.

This other way is what Adamczyk calls "closure" modeling. This is not to be confused with classical turbulence modeling, although the form of the closure terms looks very similar to turbulence terms and, in fact, they include stochastic turbulence as one component. While this method requires very complex physical modeling, it offers a sound rationale understanding of the time averages and allows a systematic evaluation of the importance of the various unsteady terms. An example of a multistage computation with this method is shown in figure 12, the total temperature distribution in the SSME fuel pump turbine. Of particular significance for heat transfer, one can see a very strong mixing and redistribution of the initial total temperature profile. This is the "free-stream" boundary condition for the convective heat transfer and, if we cannot get that right, we surely will not get the heat transfer right.

The thoughts expressed in this section can best be summarized by looking at a chart (fig. 13) of the hierarchy of experimental and computational tools at our disposal today. As one climbs the stairs illustrated in figure 13, one gains increasing information and detail at the expense of increased computational and experimental complexity, time and money. Going down the stairs yields simplicity and the broader picture (which is what the designer really wants and needs) at the expense of loss of information which must be gotten elsewhere. The challenge is balance and the key is a rational methodology

which allows us to move from block to block intelligently. The shaded areas are the ones receiving the most attention today.

As was pointed out earlier, the authors intended to concentrate on the unsteady nature of the turbine and the time averaging ideas; however, there are many other important heat transfer problems in turbines and they deserve some mention.

First, not all important heat transfer areas are subject to unsteady external flows, although they may contain embedded unsteadiness, such as separation and turbulence. One such example is the endwall region in the first vane row, which sees the very hot gases from the combustor. Turbine endwall regions show very complex heat transfer patterns, as illustrated in figure 14. These data were acquired by Hippensteele and Russell (1988), using liquid crystal technology in a large plexiglass linear cascade. Figure 14(a) is the actual liquid crystal isotherm (also iso-heat transfer coefficient) and figure 14(b) is a color graphic composite of the data. These contours suggest that the phenomenon is highly three-dimensional and will require a full three-dimensional analysis. These data and recent data by Boyle and Russell (1988) show that the shape of these contours is strongly dependent on Reynolds number, which further complicates the problem. Full three-dimensional Navier-Stokes codes, such as one by Chima (1988), illustrated in figure 15 are under development in many places. They show good promise for understanding these flows, but to date very few of them calculate heat transfer with much accuracy, if at all.

Although full closure modeling in rotating machines requires averaging in several time frames, there is still an important role for "turbulence modeling" as most people understand it. In order to properly evaluate wall related phenomena, either heat transfer or skin friction, the analyses must have a proper accounting for turbulence effects. The flows in propulsion systems are subject to high levels of free-stream turbulence and are frequently at Reynolds numbers in the laminar-turbulent transition region. Thus, turbulence and transition modeling go hand in hand in turbine heat transfer. The transition associated with such flows is largely nonlinear and is frequently referred to as the "bypass" mode (i.e., bypassing linear stability). The mechanism of "bypass" transition is not well understood, making the use of turbulence models in this region an educated guess.

Turbulence models for describing transition and used to predict turbine blade heat transfer and friction factor have included zero-equation (mixing length), one-equation (kinetic energy equation and algebraic length scale) and two-equation ($k - \epsilon$ equations). Two-equation models do a good job of simulating the transition caused by the transport of turbulence from the free stream into the boundary layer. Rodi and Scheuerer (1985) and Schmidt and Patankar (1987) are examples of their use. Typical of these analyses, they do very well against standard benchmark test data such as shown in figure 16(a), using the data of Blair (1983). However, as soon one moves close to the turbine case, such as shown in figure 16(b), using the data of Hylton et al. (1983), discrepancies begin to appear. In addition to a general lack of knowledge of bypass transition, there is also a lack of sufficient information on the effect of curvature, roughness, pressure gradient, etc. To produce the results shown in figure 16, Schmidt and Patankar (1987) determined that a modification was required in the k equation for the turbulent production term $\nu_t (\partial u / \partial y)^2$. This was modified by a model using experimental constants. The production of

turbulence in transition is a key unknown. The two-equation model is sensitive to the initial starting profiles (k and ϵ). All this suggests that two-equation models "mimic" a transition process; they do not contain the physics.

Vancoillie and Dick (1988) state that the reason conventional turbulence models fail to give a good description of transitional boundary layers is that intermittent flow cannot be modeled by global time averages. They use conditioned continuity, momentum, and turbulence equations to describe the intermittent flow in transition. The transition region is well predicted; however, a knowledge of the transition length is required. There is a need to continue the search for turbulence models more applicable to bypass transition for use in predictive codes.

An area of special concern in turbine hot section heat transfer is film cooling. Film cooling is not confined to turbines. It is also important in combustors, augmentors, and nozzles. Film cooling is the primary method whereby turbine engine hot section parts are protected from the hostile thermal environment. Every manufacturer has an aggressive program in this area. Film cooling design methods are built largely on an empirical base with correlations based largely on very specific experiments for both hole shape and position pattern. This is not to say that there are not any very fine general datasets in the literature; there are. One rather extensive, and quite realistic, set was generated by Allison Gas Turbine under the NASA HOST Project. (See Gladden and Simoneau (1988) for a description of the HOST Turbine Heat Transfer program.) A sample of these data, reported by Hylton et al. (1983, 1988), is shown in figure 17. The figure shows a progression from no film cooling, to cooling in the leading edge region only, to, finally, film cooling over a major part of the airfoil. It is clear from these that to get optimum effect requires considerable tailoring, something hard to do a priori from analysis. Some analytic progress has been made by modifying boundary-layer codes, as is also shown on figure 17, but these are highly sensitive to experimental input. This is a critical heat transfer area that cries out for a solid analytic base. There are, of course, many fine efforts at establishing a solid base. The work of Simon et al. (1985, 1986) is but one example. Film cooling is especially unique from an analytic perspective. It is a highly localized phenomenon that must be integrated into a global flow code, that already has more than it can handle. It is both an internal and external flow problem, relative to the blade. The coolant flow affects not only heat transfer but also the aerodynamic and thermodynamic performance of the machine. The ability to efficiently and effectively film cool turbine blades continues to be an important technology in the push to increase engine performance.

While space does not allow any in-depth treatment of heat transfer in the internal passages of turbine airfoils, a few remarks are in order. First, the flow and heat transfer is very complex. A modern turbine airfoil has complex serpentine flow passages with ribs on the walls, impingement jets, pin fins, and of course, film cooling. There usually several flow splits within the airfoil, as well as the flow outward by film cooling. The passages are all short and thus can be considered entrance regions. They are subject to rotation and, consequently, strong buoyancy forces and coriolis forces. The state-of-the-art, as with film cooling, is a heavy empirical base built on specific configurations unique and proprietary to the engine manufacturers. The flows are fundamentally elliptic and thus require very large computer memories to deal

with these complex shapes. The area of turbine internal passage heat transfer relies heavily on correlations and will for sometime to come. One particularly nice generic effort, again done under HOST, (see Gladden and Simoneau (1988)), is research on internal passages with and without turbulating ribs and subject to rotation, which was done at Pratt & Whitney/United Technologies Research Center and is illustrated in figure 18. The complexity is obvious. Particularly remarkable is the variation from one surface to the other in the same passage. The heat transfer can go from an augmentation of a factor of three on one wall to no augmentation on the opposite wall. Data and correlations such as these are very valuable in the heat transfer design.

Because our heat transfer analysis capability is still quite imperfect and heavily reliant on correlation and empirical data, experimentation remains very important. Most of the above results are from laboratory simulations, albeit sometimes very realistic. Designers are always seeking data from "real engine" environments. For heat transfer this is quite a challenge. It was a major activity under HOST, as described by Englund and Seasholtz (1988). Developing such instrumentation and, then, successfully using it is very important to establishing a reliable link between the laboratory and the real world. One such example is reported by Gladden and Proctor (1985) and shown in figure 19. In a real engine environment cascade the fuel flow to the combustor was cycled over a 200 K variation at a 1500 K mean. This fluctuation was measured with a dual element dynamic gas temperature probe developed by Elmore et al. (1984) for HOST. The resulting influence on the turbine vane was measured with thin film thermocouples and translated into heat flux by a dynamic conduction analysis. The uncertainty bands are quite high at this point but the system shows real promise and research is continuing. Another example are real time, actual engine turbine blade temperatures measured by Pratt & Whitney (Przirembel (1988)) with a scanning interferometer and computer system and shown in figure 20. These data are rated accurate to 10 K in an engine operating nominally at 1700 K.

A very special area, looming on the immediate horizon for turbine engine hot sections, is the challenge of new materials - the metal matrix and ceramic matrix composites. The composites offer the highly desirable features of light weight, high strength, and, especially important for the hot section, higher melting points. We really have just begun to think of the heat transfer problems. One example will be discussed in the section on HYPERSONICS. For now just a couple of remarks are in order. The early great hope was uncooled ceramic matrix composite or carbon/carbon hot parts. The question becomes what if these new materials need to be actively cooled? What do we know of the heat transfer of composites? For example, a good composite has an "imperfect" bond between the fiber and the matrix. The bond must be good enough to transfer the load among the fibers yet so not good that it breaks the fibers. The thermo-physical behavior of such a system is presently unknown. The heat transfer of high temperature composites is a new horizon.

In summary, the turbine engine remains the backbone of subsonic/supersonic flight. In order to achieve its maximum potential, designers are pushing the hot section to higher and higher temperatures - approaching stoichiometric. This represents a major challenge to heat transfer engineers. The computational and experimental tools available today are truly impressive. The ability to compute flows through the engine gas path is growing steadily; however, the ability to compute the heat transfer - a wall phenomenon - is not so

impressive. Turbulence modeling in these environments has a long way to go. Furthermore, the key heat transfer component, the turbine, is fundamentally unsteady and the questions of time resolution, time averaging and the resulting closure modeling need much work. Much of the work, such as film cooling and internal passages, is highly empirical and configuration specific. Linking the laboratory to the real engine remains a major challenge and the advent of new composite materials opens even newer horizons.

SPACE PROPULSION

Space propulsion is by rocket. Rocket propulsion is usually subdivided into liquid and solid. With stable binders solids are storable, immediate response, generally reliable, but low Isp. Some liquid propellants, such as hydrazine, are considered storable, others, such as hydrogen/oxygen, are not. Most transatmospheric/space and nearly all space missions rely on LH2-LOX because of its high performance (Isp). This presents storage problems on and near the vicinity of the launch pad, and in space, particularly for lunar and planetary missions, which will be discussed later.

Much of NASA's current effort in liquid rocketry is directed at the Space Shuttle Main Engine (SSME). Conceived as an off the shelf assembly, the performance demands exceeded all known engineering designs and SSME has performed well. Unfortunately, this often lead to premature component failures. Encapsulated by a 7.5 ft diameter by 14 ft long cylinder, this rocket engine at full power produces over 500 000 lb thrust with a performance $I_{sp} = 452.9$ sec at altitude. Further the system can be throttled to 65 percent, is capable of multiple starts and weighs in at 7004 lb. Compare it to an atmospheric propulsion system, at 50 000 lb thrust, which might be about the same size and weight. The heat transfer engineers knowledge must span cryogenics to combustion gases with all the attendant regimes associated with startup and shutdown transients and thermophysical property variations. The engineer could easily spend several lifetimes and still not understand the problems. The SSME powerhead, shown in cross-section in figure 21, and the accompanying flow schematic in figure 22, are quite useful toward understanding the physical components and flow paths.

It is useful to start the discussion of this system with the overall one-dimensional systems analysis, and then some diverse heat transfer requirements such as LOX-pump bearings and heat exchanger; fuel-pump preburner; injector; and combustion chamber. Even though sophisticated technology, such as computational fluid dynamics (CFD) and comprehensive experimental research, has been applied to the components of jet and rocket propulsion systems, details of their operation, especially as a system, are unavailable; they just work. These propulsion systems, like many others, are so complex, (figs. 21 and 22) that even today no one model or assembly of models is available to provide detailed information as to their performance. For example, the one-dimensional SSME powerbalance models are used to describe what otherwise cannot yet be quantified.

The powerbalance models are lumped one-dimensionally, partially empirical, partially thermodynamic, and partially analytic models with several black boxes and adjustable constants to emulate engine performance. The entire engine is controlled by only two components, the oxidizer valves on the preburner and

fuel flow meter, (figs. 21 and 22). Thus, fixing any one item of the model precipitates adjustments throughout this fine-tuned engine. Flowrates, pressures, temperatures, and in general all the pumping requirements are functions of the SSME system - yet are not quantified except through the powerbalance model. As to heat transfer considerations in these systems models, empirical lumped modeling is often introduced into more sophisticated codes to provide insight to the transport of energy but that is as far as it goes. Unfortunately, very little heat transfer detail can be provided from these systems analyses to a turbopump manufacturer as to the hot gas generation to drive the turbine or the cryogenic nature of the pump interconnected by a short common shaft.

Without qualification, the SSME is a very complicated heat exchanger. Let us look at some examples. Referring to figures 21 and 22, the LOX pump cooling path starts by passing LOX through a hole in the center of the impeller shaft. The low pressure LOX circulates along the shaft into the preburner gas distribution dome. The preburners for both the fuel and oxidizer pump turbines are of the same design as an internal rocket engine with injectors and combustors that are protected against radial and circumferential vibration modes by baffles. This in turn provides a periodic circumferential gas temperature distribution of some 100 °F difference entering the turbine. There is also a radial temperature distribution, as shown in computations by VanOverbeke and Claus (1986) (fig. 23). It should be noted similar problems occur in both fuel and oxidizer pumps. Some LOX coolant is bled to cool the turbine bearings and seals. Leakage down the backside of the impeller and through the shaft seal cools both the seal and the pump bearing. A complex CFD thermohydraulic analysis of a LOX ball/race interface (fig. 24) performed by Tam et al. (1987) illustrates that temperature differences cause severe cyclic thermomechanical loadings that eventually lead to bearing failure. Coolant also flows along the turbine disc to the blade roots, and eventually enters the exhaust stream that enters the LOX-GOX heat exchanger. In the upper shroud ring, hydrogen enters via orifices around the upper ring with a multiplicity of connected tubes that feeds coolant to the shroud that protects the pump housing from the hot gas turnaround duct; some dumps into the chamber that cools the shroud and eventually all winds up in the exhaust stream to the LOX-GOX heat exchanger (fig. 25).

The LOX-GOX heat exchanger consists of a single 0.19 ID tube bifurcated to provide 25.8 ft of 0.325 ID 316 SS coiled tubing. Joint integrity is critical. While the exchanger leakage is extremely low, there are safety concerns and a redesign is under consideration. High pressure LOX enters the exchanger assembly, is heated in crossflow by high pressure, high temperature preburner exhaust gases and exits into the GOX outlet manifold. Thermal control is provided through a bypass system that produces a showerhead spray of LOX into the manifold. The GOX produced is used for tank pressurization and the pogo accumulator, see flow schematic figure 22. Another major concern is that the turbine exhaust gases circulating through the exchanger do not become low enough in temperature to freeze up or shed ice into the transfer ducts and the injector. As a result, the lower throttle limit is 65 percent.

The main injector is also quite a complex heat exchanger, figure 26. Baffles attenuate direct gas flows from the transfer manifolds that are maldistributed. Of the three hot gas ducts to the injector, the outer two carry the

gases, with the inner and largest duct virtually flowless; maybe even backflowing (fig. 21). Fluid hydrogen from the regeneratively cooled combustion chamber provides the enthalpy drop to drive the primary pump turbine and cools the powerhead ducts and bowls; it then flows into the injector between the primary and secondary face plates (fig. 22). Hot hydrogen rich gases from the LOX/GOX exchanger and the fuel pump turbines exhaust are impinged onto the LOX posts that are protected against vibration by vortex shedding spiral trips. An analysis of the flow over the LOX posts in the main injector was carried out by Rogers et al. (1986) at NASA Ames. It was directed at understanding forced vibration of the posts. The complex flow patterns are illustrated in figure 27. To date heat transfer has not been incorporated into these analyses but the potential for these codes for contributing to the heat transfer analysis is quite impressive. These hot gases heat the LOX which is flowing from the upper LOX plenum into the combustion chamber.

The main problem on the coolant side of these heat exchangers is the tremendous variation of thermophysical properties, particularly the density. Hendricks (1979, 1980) has shown that a vast array of data can be correlated by relating the data to the fluid compressibility and by using corresponding states type concepts from thermodynamics. An example of such a correlation is shown in figure 28. Similar to the situation with the turbine engine, which was discussed above, advanced computational tools are being brought to bear on the gas path flows, while the internal passages still rely largely on correlation.

The main combustion chamber is really a combustor and nozzle combination (fig. 21). In addition to the severe cooling requirements at the throat and the constraint that $T_{wall} < 1000^\circ\text{F}$, the liner thermomechanical cycle exhibits 2.5 percent strain (with 0.2 considered yield) i.e., operates in the plastic zone. It must deform without cracking or buckling during startup and run, and it must reform upon shutdown without failures. Many structural failures are instigated by thermocycling components that are not thermocompliant. Thermocompliant systems deform with changes in temperature and flow passages must conform to take advantage of surface alterations. The strains introduced by welds or rigid surfaces can be severe and give way to fatigue cracking. The turbine inlet shroud that diverts hydrogen rich combustor gas to the turbine could be rigid or fabricated of materials that deform to mitigate thermal shock. Turbine blade temperatures are limited by materials and coatings are not all that thermocompliant or reliable. Failures are most often due to excessive compressive stress that causes the coating to spall or break free.

As with the turbine engine, health monitoring in flight and research instrumentation of rocket propulsion systems is very important. In addition to the severe operating environment an instrument failure offers the potential for opening a path between the oxygen and hydrogen. Thus, a major problem plaguing designers and researchers alike is sensor failure.

Space storage and transfer of cryogenic propellants whether near the vicinity of the launch pad, on the space station, and in particular for lunar and planetary missions, pose unusual heat transfer and materials problems. Missions such as space station and beyond will require large propellant storage depots and for some missions propellant production on the plants or their moons have been given serious consideration. For subcritical propellant storage on

Earth one usually knows where the liquid is; but in space (or bases with microgravity) propellant orientation is a serious concern. In addition heat leaks from support structure, fittings, insulation, and degradation due to impalement of space debris represent challenges to long term storage of cryogenics in space. In microgravity, local nonuniform sources of heat usually disturb the average void/liquid fraction which, along with the energy of the container surface, determines how the liquid will be distributed. One cannot afford to vent liquid, nor permit a gaseous pocket to become over heated posing the threat of fluid rollover causing a sudden uncontrollable rise in pressure. In many cases separated fluids can be mixed by shaking, so one method of controlling propellant distribution is to use small thrusters to create pulsed settling, i.e., create a small g-field. The challenge is to determine the proper frequency, amplitude, and duration. CFD results by Hochstein (1987) (fig. 29) illustrate propellant distribution response in a tank subjected to 0.008 g at 0.1 Hz for 0.1 sec duration. Propellant orientation is necessary in a long term storage regimen, but more importantly to microgravity transfer of propellants where it is important to transfer liquid. Transfer can be assisted by fluid rifling, i.e., using a portion of the axial momentum for angular momentum thereby orienting the propellant and permitting its lighter components to go to a liquifier/expander-heat exchanger prior to vent. But venting is a last resort.

Most of the above has been directed at the low Earth orbit SSME; however, the long term goal is to explore the far regions of the solar system. Over the years, results for minimum orbital transfer energies along with propulsion systems as ion, arc, and beaming have been studied and assessed. Minimum energy trips to Mars take a long time (3 years) and usually implies a nuclear power source (0.1 to 1 MWe), and ion thrusters which have very low thrust but high performance (Isp to 5000 at 50 cm diameter). They are often considered for interplanetary and extra solar activities. Such an application might be a cargo transfer vehicle. However, if mission transfer times are to be reasonably short (months) then chemical rocket propulsion appears necessary.

The return from deep space or high Earth orbit requires some type of slow-down or braking. One approach is to use the atmosphere as a brake, as shown in figure 30. In the aeroassisted orbital transfer vehicle, the propulsion system and the vehicle must be combined to provide transfer from higher to lower orbits. In hypersonic reentry, the forward body of the vehicle becomes very hot. If flow about the body is not stable, heat can be directed into the payload or, even worse, an unscheduled reentry can take place. The subject of aerodynamic heating will be discussed in the next section.

In summary, the backbone of space propulsion is the chemical rocket engine. It has numerous severe heat transfer problems. In general the heat transfer and fluid flow technology for rocket propulsion currently lags that for turbine engines. As with turbine engines, the tendency in performing fluid flow and heat transfer analyses is to use advanced CFD codes on the gas path side and correlations on the coolant side. A key to advancing the heat transfer technology will be developing the ability to understand and predict cryogenic fluid behavior with the same level of sophistication as is done with gases. The Space Shuttle was built largely on 1960's technology. That is all changing rapidly. The space station, the Civil Space Technology Initiative (CSTI) and the Mars mission (Project Pathfinder) are all demanding advanced technology.

HYPERSONIC/TRANSATMOSPHERIC PROPULSION

The flight regime of vehicles, such as the National Aerospace Plane (NASP), which will carry one from runway to orbit is particularly exciting, because it is the true marriage between aero- and space- to form aerospace flight. Several aeropropulsion concepts are possible for the transatmospheric flight regime. These concepts cover a range of operating cycles from the turbojet/turbofan to the ramjet/scramjet. Since the lower-speed flight regime (turbojet/turbofan) and the upper end of the flight regime (rockets) have been discussed, this section will be concerned with the intervening regime (i.e., hypersonics). The transition from supersonic to hypersonic flight is somewhat ill-defined but is generally considered to occur in the Mach number range of 4 to 6. This transition does not represent physical phenomena, such as that which occurs at Mach one, but is more of a convenient reference. The airturbo-ramjet, shown in figure 31, is capable of operation in the Mach number range of 0 to 6. When combined with a rocket this propulsion system could conceivably boost the vehicle to orbital velocity. The inclusion of a scramjet (fig. 31) (supersonic combustion) can extend the vehicle flight Mach number range to about 12 before it would be necessary to switch to a rocket to attain orbital velocity. This section will discuss some of the heat transfer challenges in accelerating from supersonic flight to orbital velocity.

Vehicle flight at hypersonic speeds presents significant heat transfer challenges throughout the propulsion system and the airframe because of the very high aerodynamic heat loads encountered. Local stagnation areas can experience heat flux levels exceeding 50 kW/cm^2 . Heat flux levels up to 10 kW/cm^2 are common throughout the propulsion system. By necessity, the engine and the airframe are highly integrated. Not only do they share a common structure at the engine/airframe interface but the airframe also acts as a compression surface for the inlet and as an expansion surface for the nozzle.

A list of heat transfer challenges for high speed flight includes: very high aerodynamic loads, laminar-turbulent transition, shock/shock and shock/boundary layer interactions, film cooling and skin friction reduction, advanced composite materials, combined thermal/structural analysis, real gas effects and wall catalysis, and thermal/management of the integrated engine/airframe environment. These challenges probably look familiar since they are major issues common to any type of advanced heat engine, i.e., management of high heat loads in order to maintain structural life and integrity. However, there is an emphasis here on high-speed flow phenomena such as shock interactions near a surface, advanced material performance, and thermal management in an integrated engine/airframe environment. This list is not meant to be inclusive but to illustrate the scope of the challenge. Each of these elements will be discussed in the following paragraphs.

Knowing where the boundary layer transition to turbulence begins and its extent is just as important at hypersonic flight speeds as it is at subsonic speeds. The momentum Reynolds number at which transition occurs is shown in figure 32 as a function of the free-stream Mach number. A "rule-of-thumb" that has been used in the past is that transition occurs when $Re/M = 300$ (momentum thickness Reynolds number over Mach number) which is the upper curve in the figure. This approach is obviously inadequate since most of the experimental data show that the boundary layer would transition much further upstream than the predicted location. Modeling of the physics of transition are required to

improve prediction capability. Mack (1975) of JPL and Hefner and Bushnell (1979) at the NASA Langley have had some success on external airfoil surfaces with linear stability theory and the n -factor approach. Demetriades et al. (1980) of Montana State University has attempted to define a sufficient and necessary condition for transition with some success over a range of Mach numbers and surface conditions, etc. However, the prediction of boundary layer transition in internal passages with supersonic flow is complicated by the addition of expansion and compression waves, weak shocks, etc.

Shock wave interaction with the boundary layer can have a significant effect on the local aerodynamic heating of a surface. The impact of an oblique shock interacting with a turbulent boundary layer in a free-stream Mach number of 4 is shown in figure 33. These data from Hayashi et al. (1984) show a substantial increase in pressure as well as the heat transfer coefficient. In addition, the measured heat transfer coefficient shows a local boundary layer separation and a subsequent augmentation of about 3:1. A comparison of experimental data for a normal shock/boundary layer interaction and a two-dimensional Navier-Stokes analysis is shown in figure 34. These data were acquired in the NASA Lewis 1-ft supersonic tunnel by means of laser anemometry. There is a good comparison of Mach number contours with the analysis in the upstream portion of the flow field. However, in the downstream region three-dimensional effects are significant and none of the flow physics is adequately represented. Analytical modeling and experimental verification of this phenomena are continuing. However, there is a significant amount of challenging work left to complete the current effort and extend the results to high enthalpy flows.

Another phenomena that occurs in high-speed flight is the shock-on-shock interaction that occurs at an engine inlet or strut leading edge as the vehicle accelerates through a particular Mach number region. The physical phenomena is reasonably well understood and is depicted in figure 35. Six classes of interaction have been identified, however, the most severe heating occurs with the type IV interaction. Wieting (1987) of the NASA Langley has measured stagnation point heat transfer augmentation factors of up to 10:1 for the type IV interference heating phenomena. Recent discussions with Wieting suggest these factors may approach 30:1 under certain conditions. A typical schlieren photograph of a type IV interference pattern is shown in figure 36. Navier-Stokes analysis has been able to capture most of the flow physics characteristics of this phenomena, however, the challenge is finding methods to reduce the heat load and/or finding cooling schemes that can tolerate or accommodate these high heat loads. These could be either active or passive cooling techniques.

Much research was conducted in the 1960's and early 1970's on film cooling in a supersonic free-stream environment. However, very little research on film cooling has taken place since that time period. Figure 37 depicts the current status of supersonic film cooling. Depending on the dataset/correlation used, substantial differences in effectiveness (and skin friction reduction) will be predicted. These differences may be attributed to variations in test facilities, however, understanding the physics and adequately representing that physics in the analytical model is a major driver. In particular, properly modeling the shear layer mixing between the film layer and the free stream could improve the prediction accuracy. In practical applications, film cooling may be the method of choice to accommodate the large heat flux augmentation that occurs in a shock-boundary layer interaction. The challenge will be to

maintain the film integrity through a local boundary layer separation that may occur at this location.

Heat pipes are a potential means of passively cooling structures exposed to high heat flux levels. These devices have been studied for several years to develop this technology and to understand their functional characteristics and their operating limits. Silverstein (1985) has shown (fig. 38) the heat flux limits of several substances. Lithium, operating at a high vapor pressure, could theoretically transport the high heat flux levels associated with shock-on-shock interference heating to a heat sink thereby protecting the structure and maintaining reasonable material temperatures. However, many questions remain regarding the fabricability and operability of the devices in a hostile environment. Material selection must be concerned about operating temperature limits, corrosion/erosion, ductibility, etc. The successful operation of the heat pipe is dependent on its "dry-out" limits, the sonic velocity of the transport medium as shown in figure 38, and its ability to respond to transient phenomena. Although research is continuing on these devices at places such as Los Alamos National Lab, demonstrating this technology in a simulated high heat flux environment presents a challenge.

The high heat flux encountered by the leading edge of a hypersonic vehicle in flight imposes severe demands on the materials and structures used for these applications. The aerodynamic heating, including the shock-on-shock discussed previously, at high flight Mach numbers creates the high heat flux with corresponding high surface temperatures which can exceed the melting point of conventional metallic and potential ceramic materials available for aerospace applications today. Not only must the high heating rates be tolerated but the distortions caused by thermal warping of the structure must be kept to a minimum to achieve high inlet performance. Consequently, there is a need to develop new materials incorporating active/passive cooling schemes able to withstand these severe environmental and thermomechanical conditions.

A combined analytical and experimental research effort has been initiated at NASA Lewis (Melis et al. (1988)) to assess the capability of actively/passively cooled structures to tolerate the high heating rates typical of hypersonic flight. In addition, materials technologies and fabrication techniques are being studied for applying advanced metal matrix and ceramic matrix composite technology to actively/passively cooled structures. This program represents an interdisciplinary approach to focus the structures, fluids, materials, design and instrumentation disciplines on the problem. This approach is represented by the flow diagram in figure 39. The final step in the loop is the experimental verification in the Hot Gas Test facility (shown in the upper right) which can provide heat flux levels up to 10 kW/cm^2 at the stagnation point. Earlier experiments on simpler shapes are reported by Melis et al. (1988) and indicate that the analytical and experimental tools being brought to bear on this problem will be most useful.

Thermal management may well be the critical element in permitting flight at hypersonic speeds. The very large aerodynamic heating loads on both the engine and airframe must be dealt with, if the vehicle is to survive. The magnitude of the problem is indicated in figure 40, taken from Anderson et al. (1987) which shows that at Mach 10 flight velocity, 80 percent of the heat capacity of flowing fuel heat is needed for thermal management. This assumes a cryogenic fuel, such as hydrogen, is being used. In addition, there must be a

balance between the heat generated, the heat absorbed, the fuel required for propulsion, and the fuel required to operate auxiliary equipment in order to get closure on the problem. The uncertainty generally associated with heat transfer (± 10 to 20 percent) is a challenge to the designer in trying to get this closure. These uncertainty levels represent a significant increase in fuel requirements to satisfy the heat sink requirements and would have a serious impact on the payload capability of the vehicle.

In summary, the transatmospheric flight envelope will probably require a hybrid engine combination. In the low hypersonic regime the airturbojet is a candidate, but for most of the hypersonic regime the scramjet is the backbone propulsion system. For heat transfer engineers the interesting feature in this regime, which is different from the two extremes already discussed, is that it is almost impossible to sharply divide where the airframe lets off and the engine begins. The heat transfer problems are very similar, whether one is talking about the engine or the airframe. Thus, the reader will note that there was very little engine specific discussion in this section in contrast to the previous sections. Prior to NASP, hypersonics had been on a 15 year hiatus. The work shown here is rather preliminary, but it is moving fast.

Two additional points should be emphasized. Progress in this flight regime is highly dependent on the computational power of advanced computers and an ability to analytically model problems and solve Navier-Stokes equations with the energy equation and real-gas effects (high temperature, reacting flows, etc.). The status of CFD for hypersonics is discussed in a recent article by Dwyer et al. (1987). Finally, experimental verification of these models is generally limited to the lower end of the flight regime. Verification of the higher flight Mach numbers is dependent on prototype vehicles.

CONCLUDING REMARKS

In this very brief review of heat transfer in aerospace propulsion we have seen areas of commonality and areas of difference. In the subsonic/supersonic propulsion arena the primary propulsion system, the turbojet/turbofan engine, is a very sophisticated machine and some very advanced computational and experimental tools are being brought to bear on achieving an optimum design. Sparked by the advanced fluid mechanics and structural analysis capabilities and the advent of exciting new materials, the industry and government have laid down a challenge to double the performance of today's engine by the turn of the century. The space program, on the other hand, has built a very sophisticated power plant largely on an empirical base. The present revitalization of the space program is calling for a significant advancement in the technological tools used to design and build space propulsion systems. Finally, in a new-old area, hypersonics, a breakthrough demanding technology of 15 years ago is being resurrected and combined with new advances to bring to reality the true aerospace vehicle, the transatmospheric vehicle.

In all of these areas the goal is to concentrate more and more energy into a smaller and smaller envelope, placing ever increasing demands on the knowledge of heat transfer. As a general rule of thumb, the heat transfer problem is harder to solve than the fluid mechanics, and the ability to accurately describe the heat transfer significantly lags the fluid mechanics.

As the ASME Heat Transfer Division enters its second half century, the field of aerospace propulsion offers it a challenge worthy of its history.

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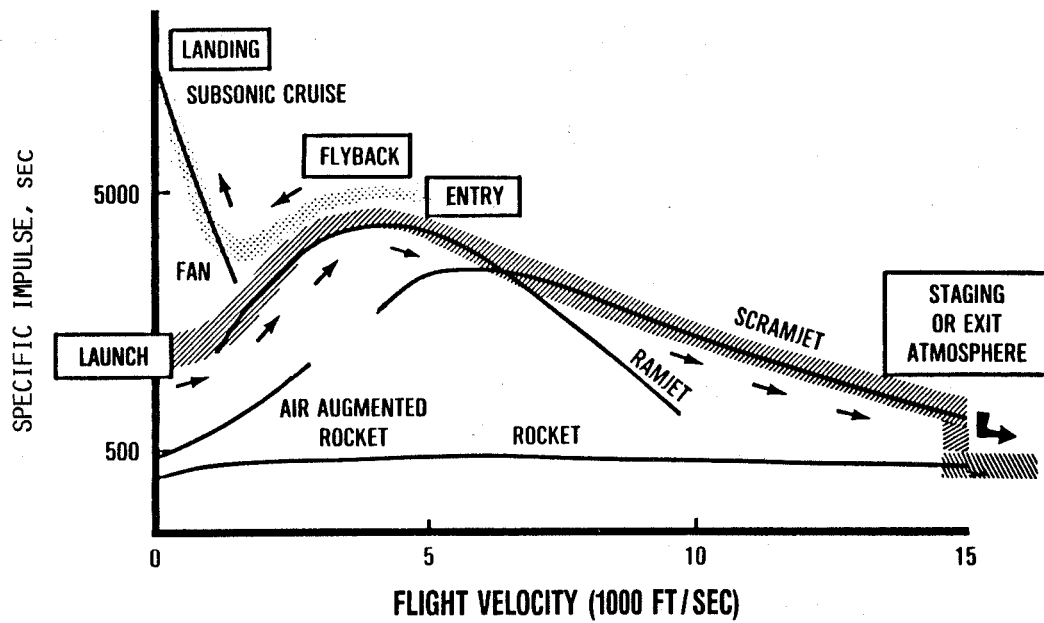


FIG. 1 HIGH MACH PROPULSION EARTH-TO-ORBIT MISSION SENARIO.

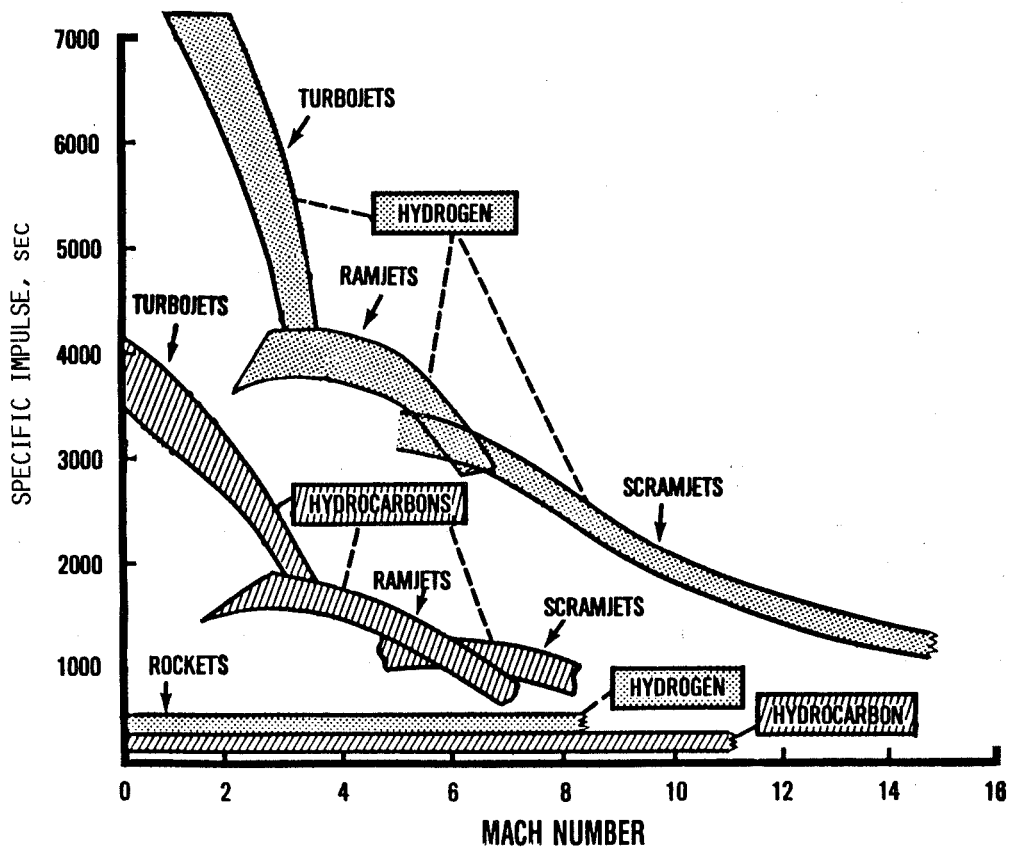


FIG. 2 PROPULSION OPTIONS IN THE HIGH MACH NUMBER FLIGHT ARENA.

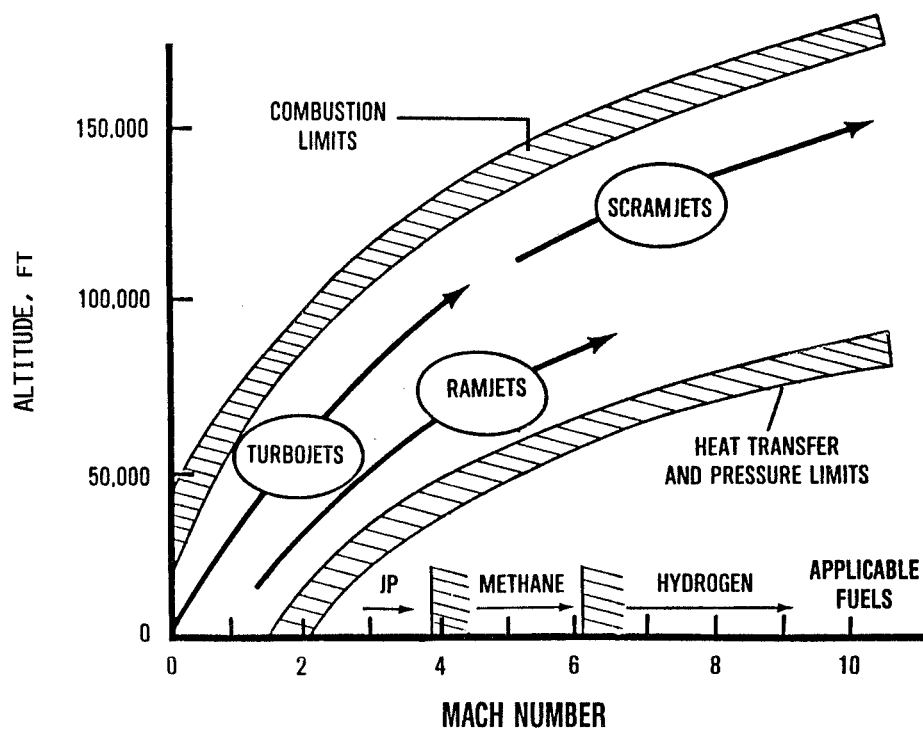


FIG. 3 OPERATING CORRIDOR FOR AIR BREATHING PROPULSION OPTIONS IN THE HIGH MACH NUMBER FLIGHT ARENA.

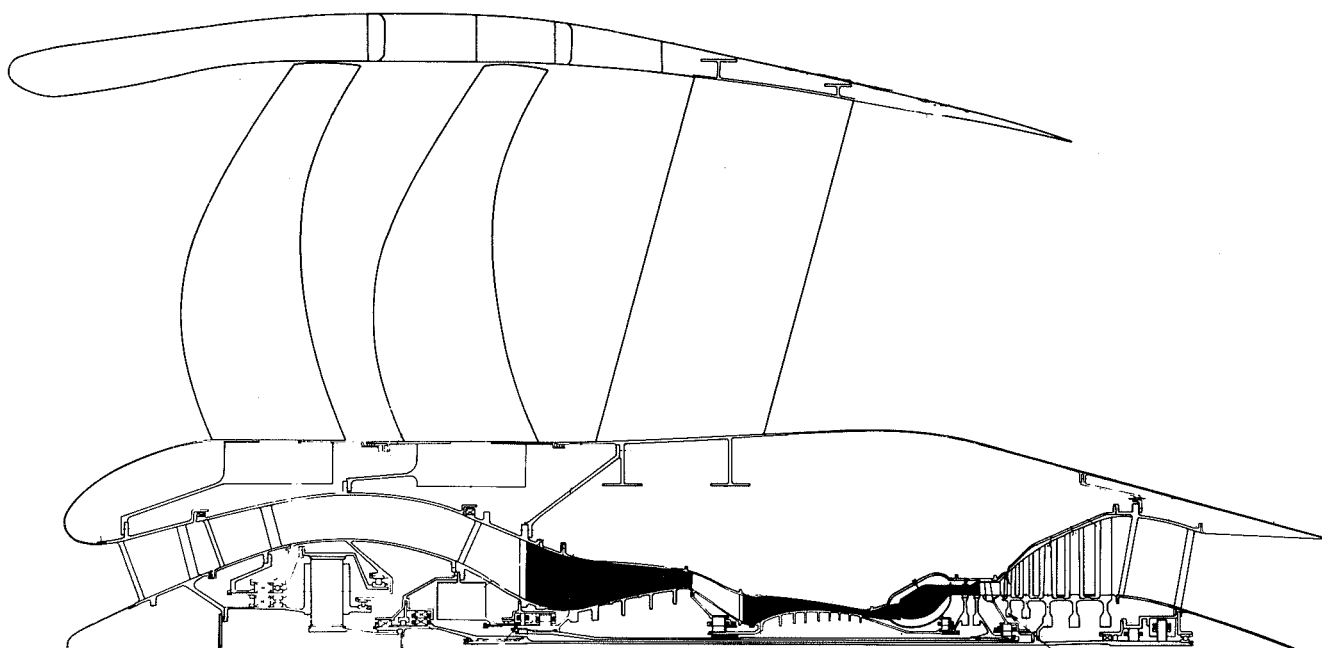


FIG. 4 CROSS-SECTION OF A CONCEPTUAL ULTRA-HIGH BY-PASS RATIO ADVANCED COMMERCIAL TURBINE ENGINE OF THE FUTURE.

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OF POOR QUALITY

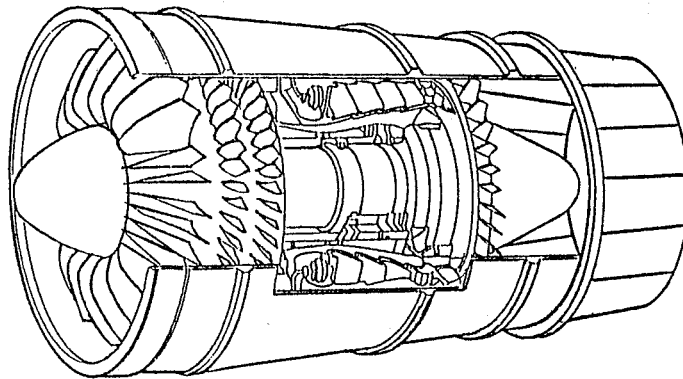


FIG. 5 ARTIST'S VIEW OF A CONCEPTUAL ADVANCE
MILITARY TURBINE ENGINE OF THE FUTURE.

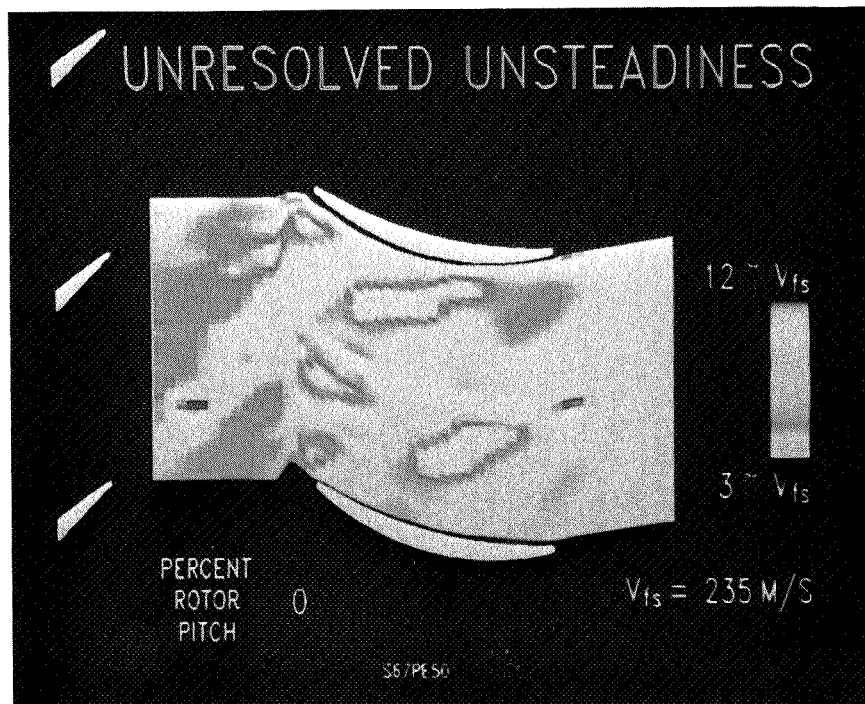


FIG. 6 DISTRIBUTION OF THE KINETIC ENERGY OF A ROTOR WAKE AS
IT PROGRESSES THROUGH THE DOWNSTREAM STATOR PASSAGE (ONE FRAME
FROM A DATA MOVIE).

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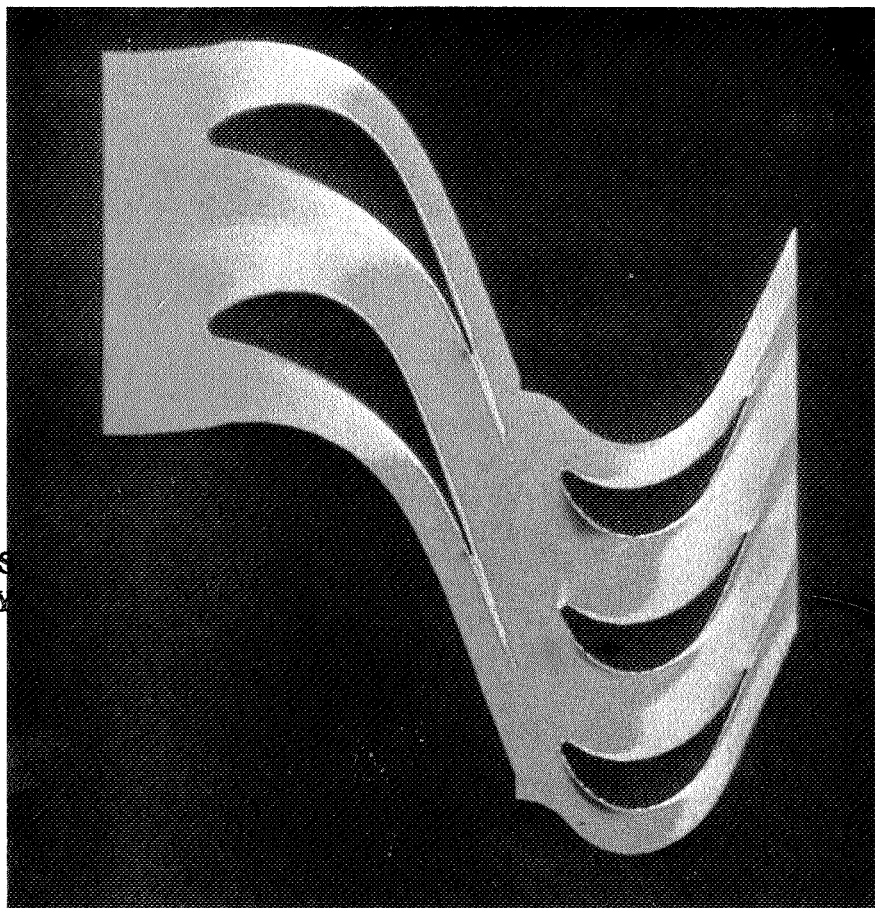


FIG. 7 TIME-ACCURATE TWO-DIMENSIONAL NAVIER-STOKES CALCULATION OF THE MACH CONTOURS IN THE SSME FUEL PUMP TURBINE (3/2, BLADE-VANE COUNT).

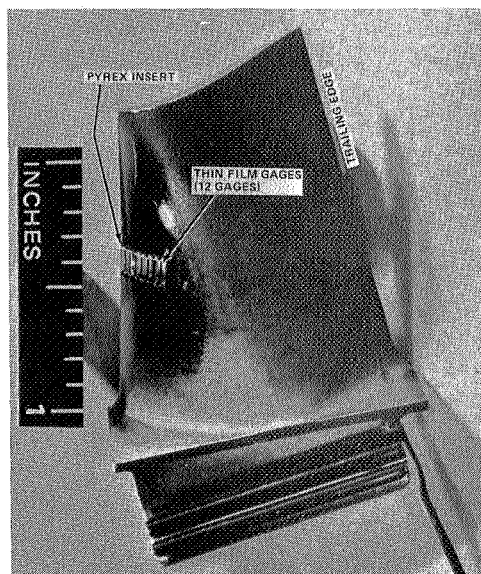


PHOTO OF A ROTOR BLADE INSTRUMENTED WITH THIN-FILM SENSORS

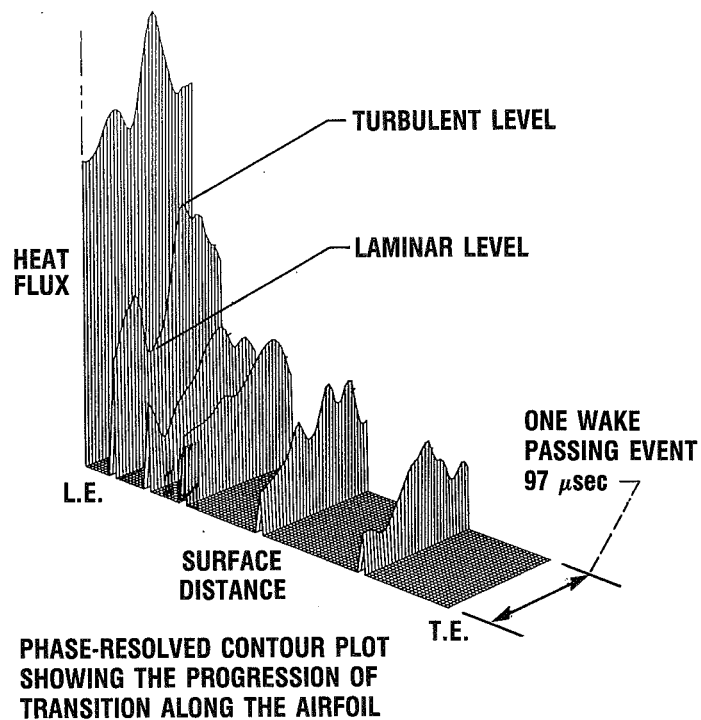
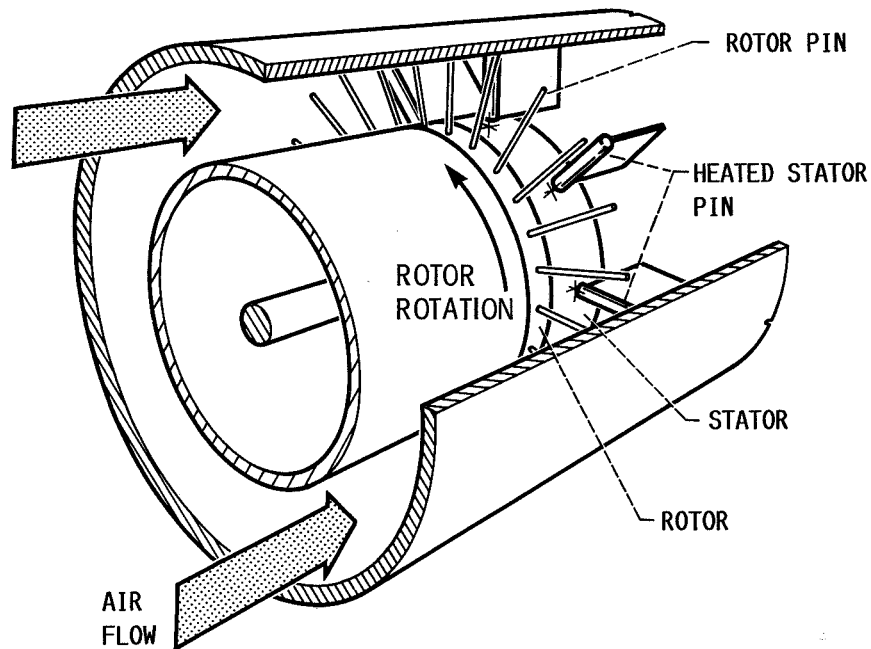
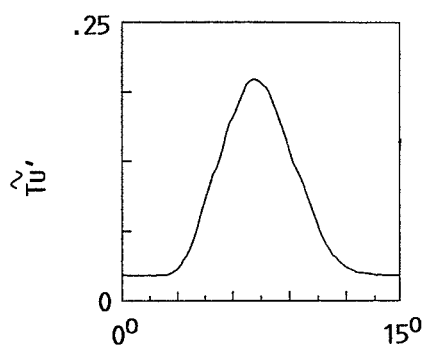


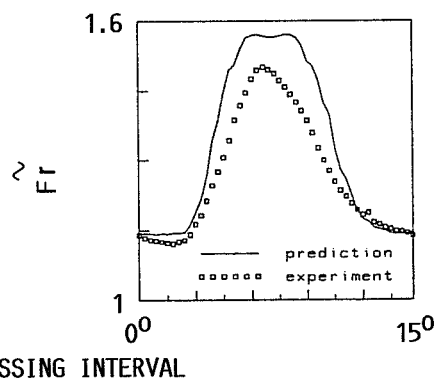
FIG. 8 EFFECT OF WAKES ON LAMINAR-TURBULENT TRANSITION IN A TURBINE STAGE.



(a) CONCEPTUAL VIEW OF EXPERIMENT.



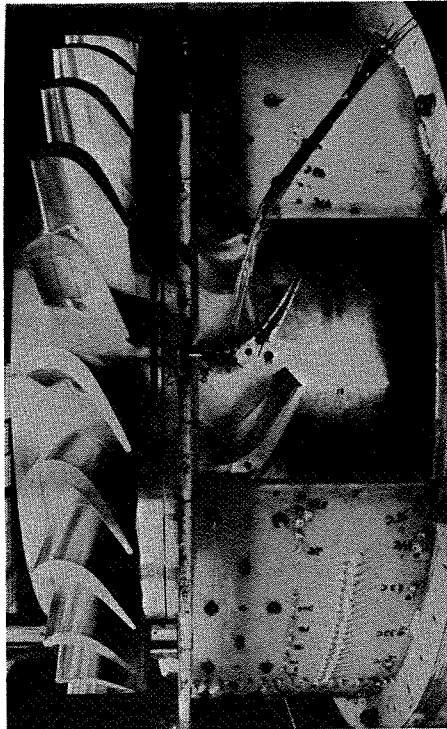
(b) PHASE-RESOLVED RMS
FLOW UNSTEADINESS.



(c) PHASE-RESOLVED HEAT
TRANSFER (FROESSLING
NUMBER.).

FIG. 9 ENSEMBLE AVERAGES OF FLOW UNSTEADINESS AND STAGNATION HEAT TRANSFER MEASURED IN A ROTOR-WAKE SIMULATION RIG.

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LARGE SCALE ROTATING RIG 1-1/2 STAGE
TURBINE CONFIGURATION FIRST VANE AND
ROTOR CASE REMOVED

HIGH REYNOLD'S NUMBER; 65-PERCENT GAP; $C_x/U_M = 0.78$

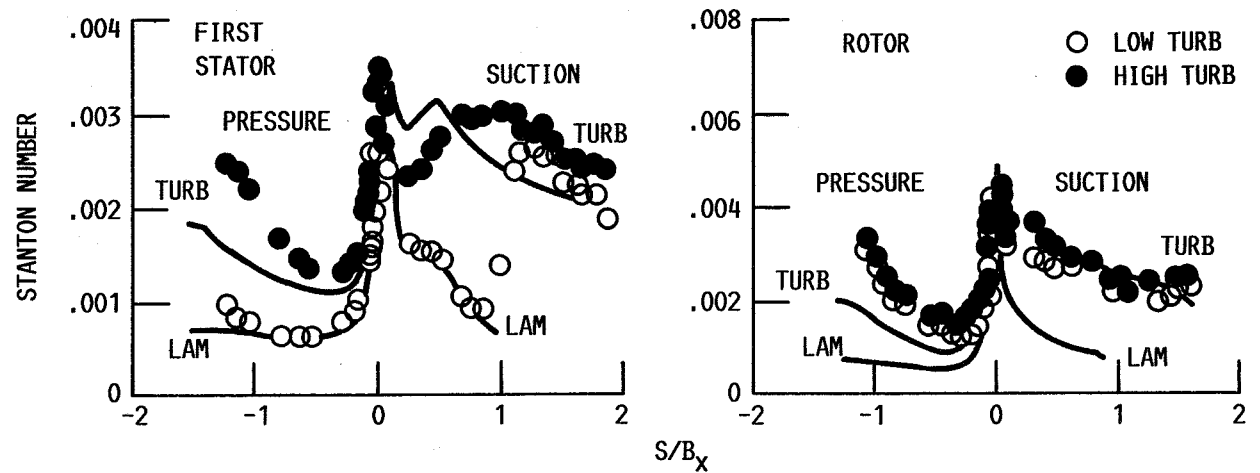


FIG. 10 EFFECTS OF BOTH INLET TURBULENCE LEVELS AND ROTOR/STATOR ON TURBINE MID-SPAN AVERAGE HEAT TRANSFER.

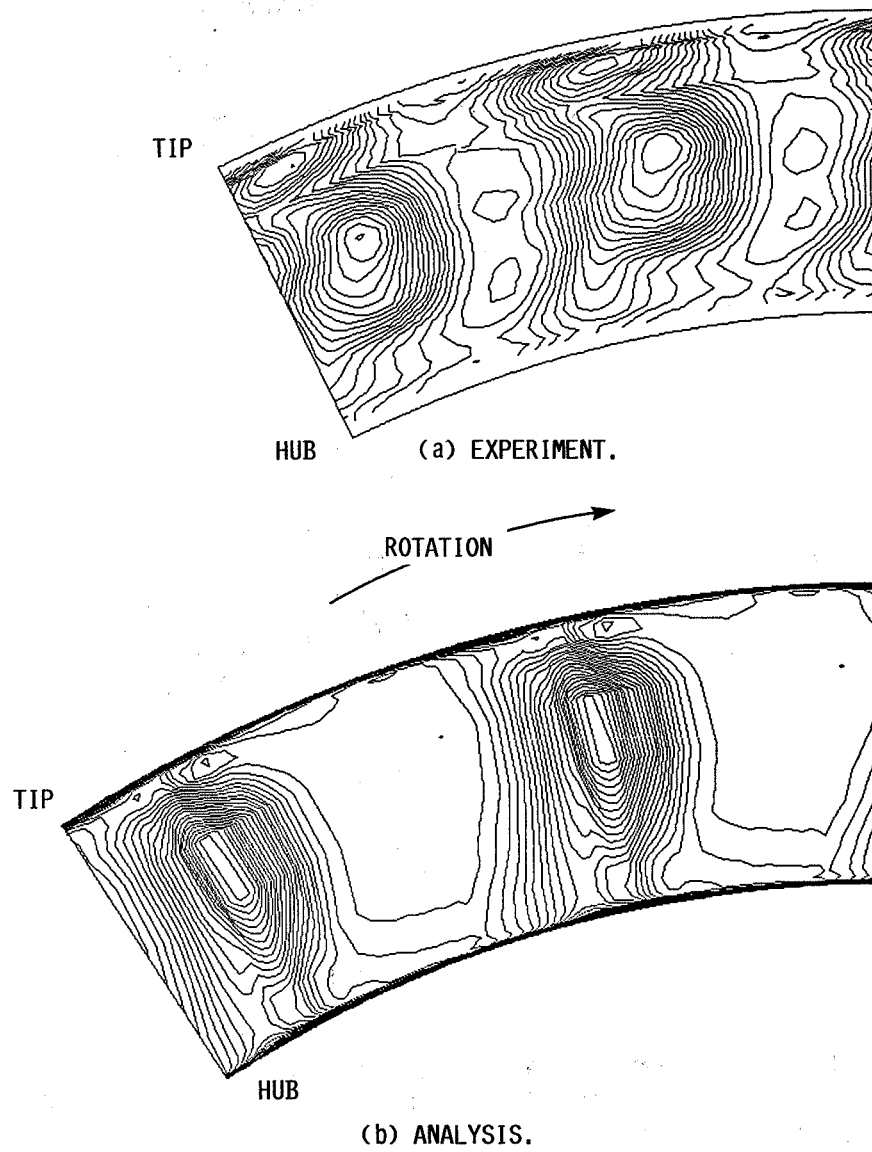


FIG. 11 RELATIVE TOTAL PRESSURE DISTRIBUTION AT FIRST STAGE ROTOR EXIT OF A LARGE LOW SPEED TURBINE.

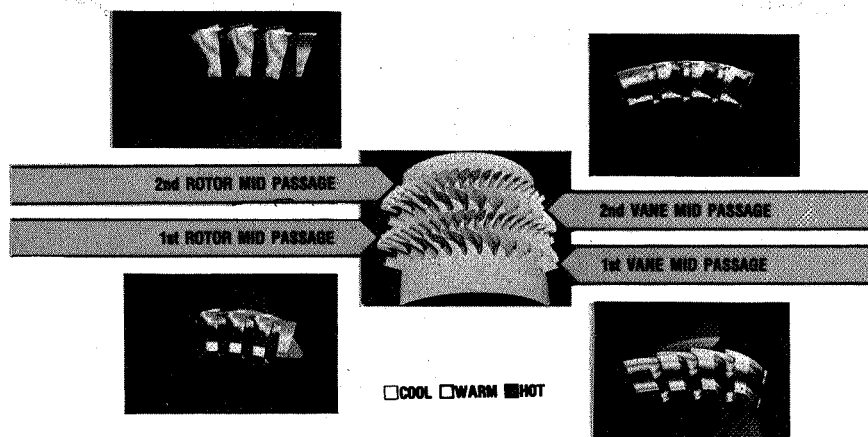
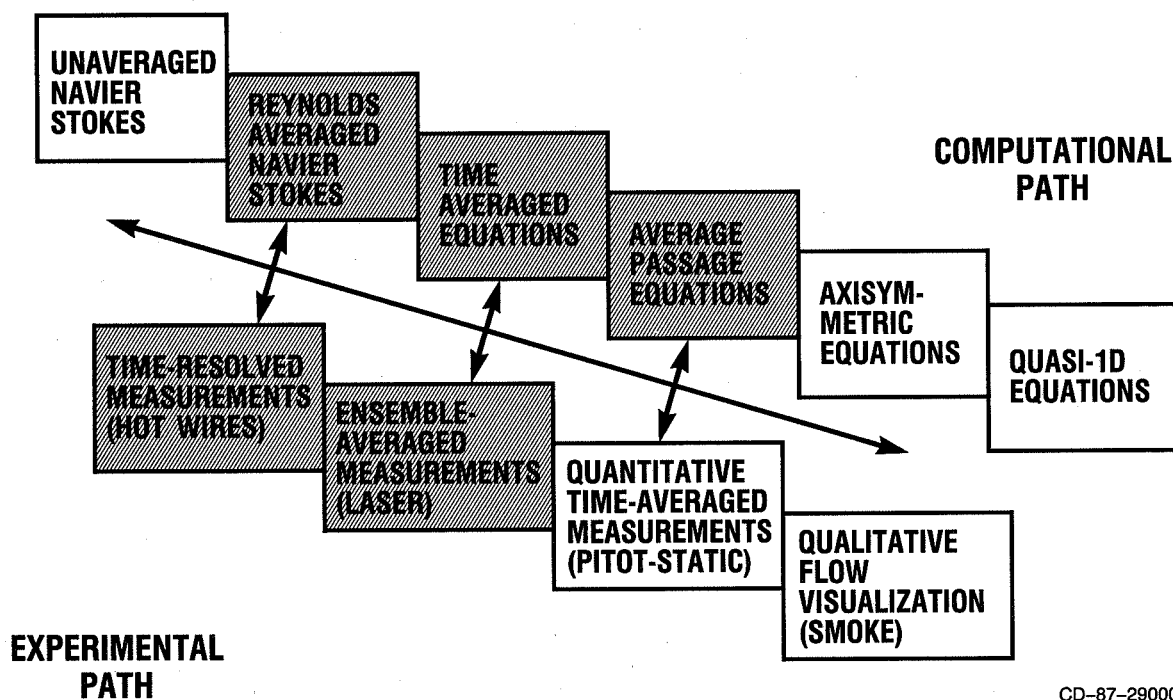
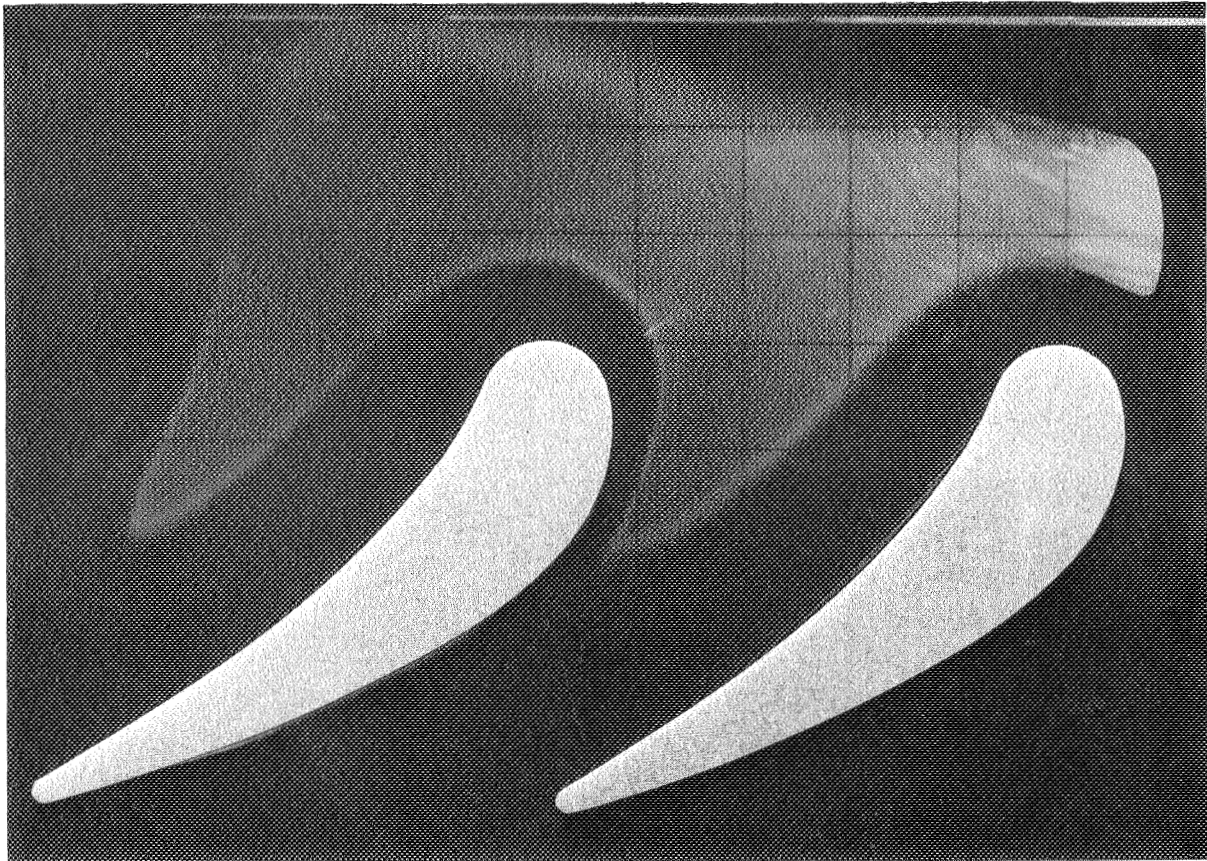


FIG. 12 COMPUTATION OF THE EVOLUTION OF THE TOTAL TEMPERATURE FIELD WITHIN THE SSME FUEL TURBINE BY THE AVERAGE PASSAGE METHOD.

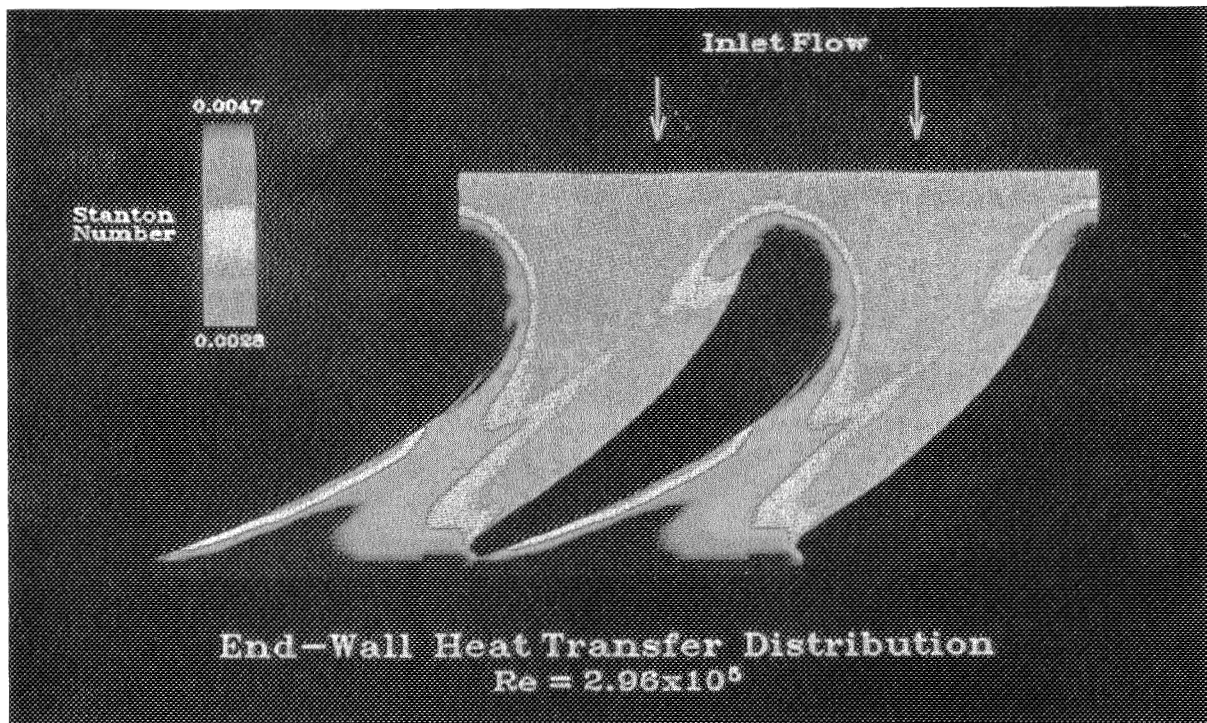


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FIG. 13 HIERARCHY OF EXPERIMENTAL AND COMPUTATIONAL TOOLS AVAILABLE FOR TURBOMACHINERY RESEARCH.

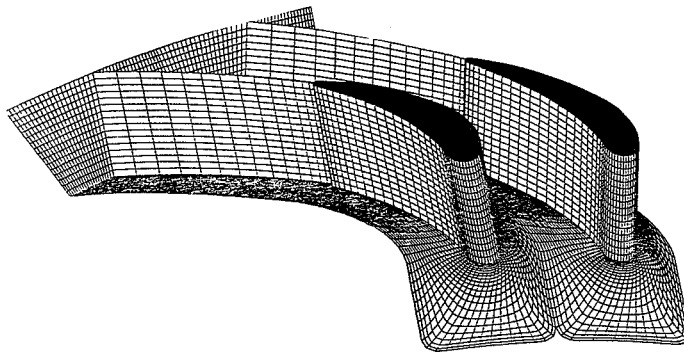


(a) EXPERIMENTAL LIQUID CRYSTAL ISO-THERM (ALSO ISO-HEAT TRANSFER COEFFICIENT).

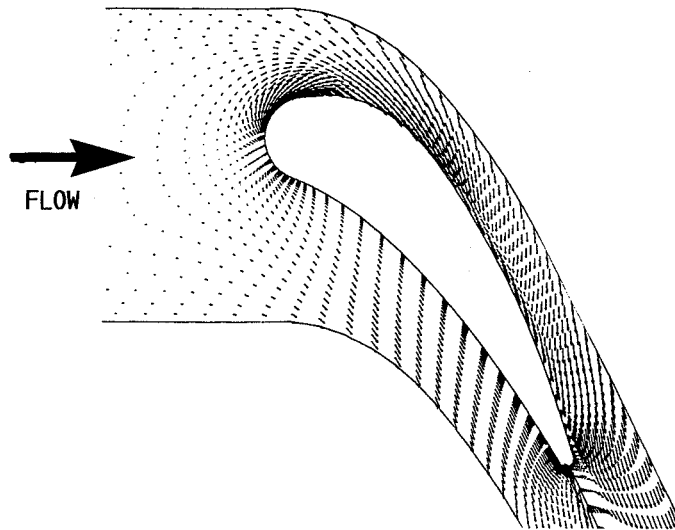


(b) COMPUTER BASED CONTOURS OF ENDWALL HEAT TRANSFER EXPERIMENTAL DISTRIBUTION.
 FIG. 14 HIGH-RESOLUTION TURBINE ENDWALL HEAT TRANSFER DATA, USING LIQUID CRYSTAL TECHNIQUES.

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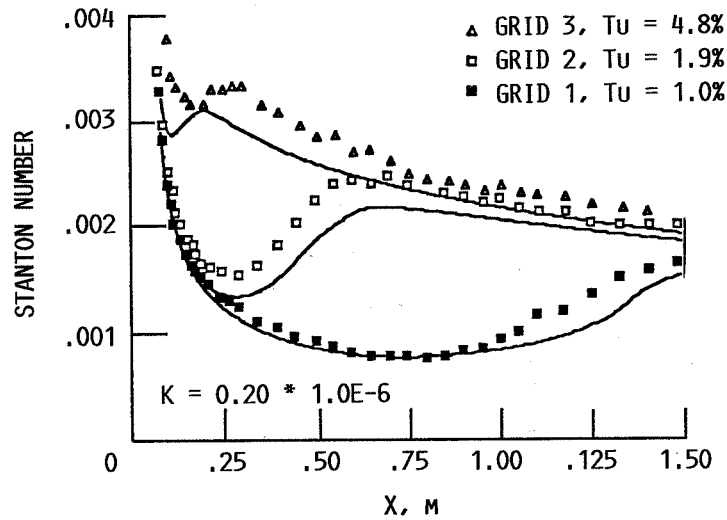


(a) GRID.

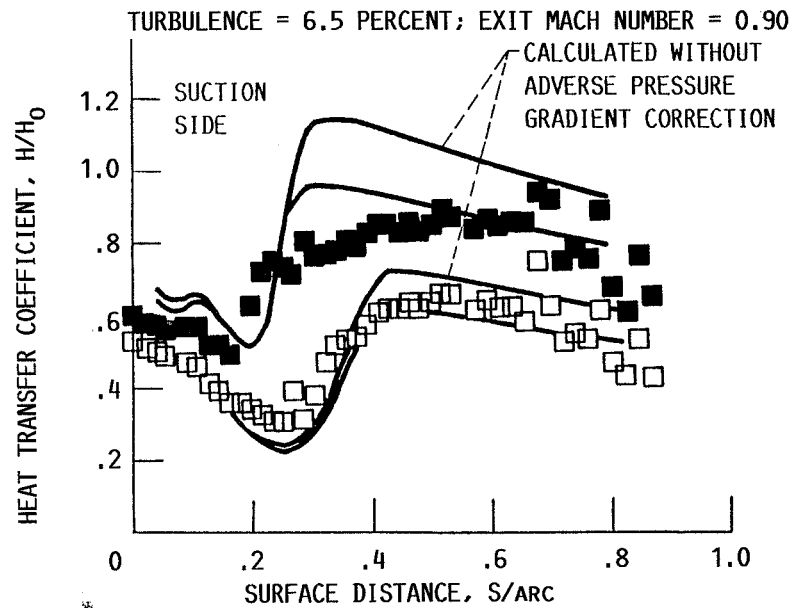


(b) FLOW VECTORS NEAR THE ENDWALL (y^+ OF 3).

FIG. 15 TURBULENT THREE-DIMENSIONAL NAVIER-STOKES TURBINE FLOW
CODE CALCULATIONS.

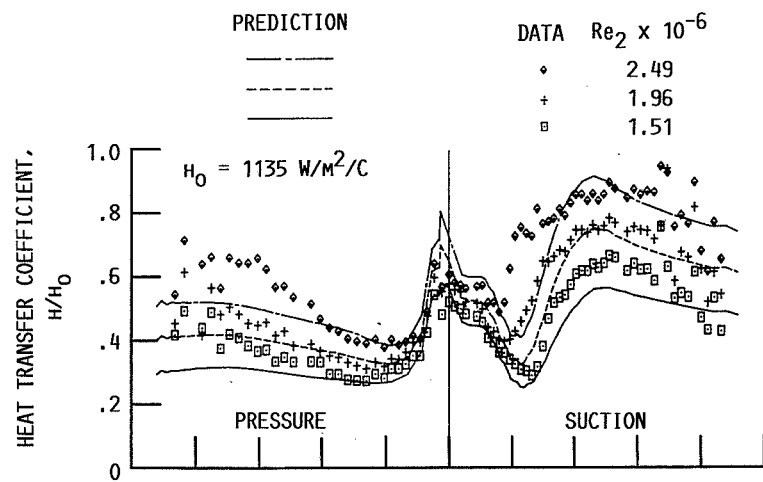


(a) FLAT PLATE SUBJECT TO MILD PRESSURE GRADIENT.

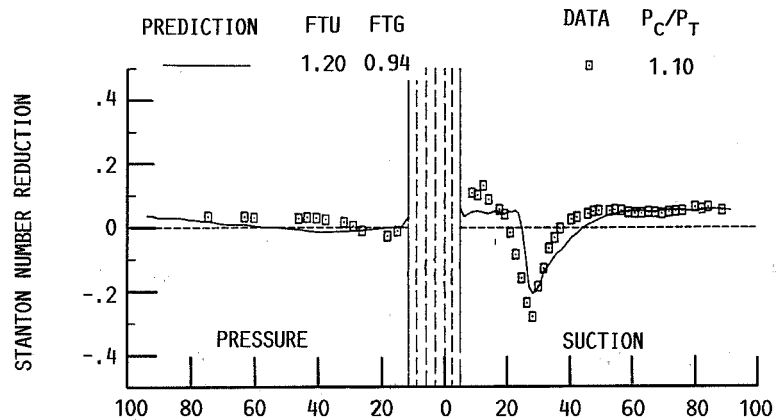


(b) SUCTION SURFACE OF HEATED VANE CASCADE.

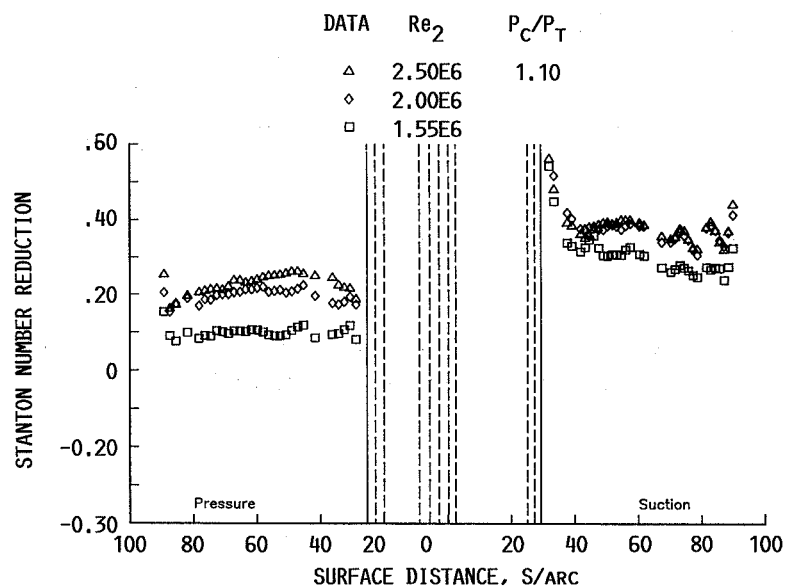
FIG. 16 COMPARISON OF A MODIFIED LOW REYNOLDS NUMBER $k-\epsilon$ TURBULENCE MODEL WITH EXPERIMENTAL HEAT TRANSFER DATA.



(a) NO FILM COOLING.



(b) FILM COOLING IN LEADING REGION ONLY.



(c) FILM COOLING IN BOTH LEADING EDGE AND DOWNSTREAM.

FIG. 17 FILM COOLING HEAT TRANSFER DATA IN A HEATED VANE CASCADE, INCLUDING SOME COMPARISONS WITH A MODIFIED BOUNDARY LAYER ANALYSIS.

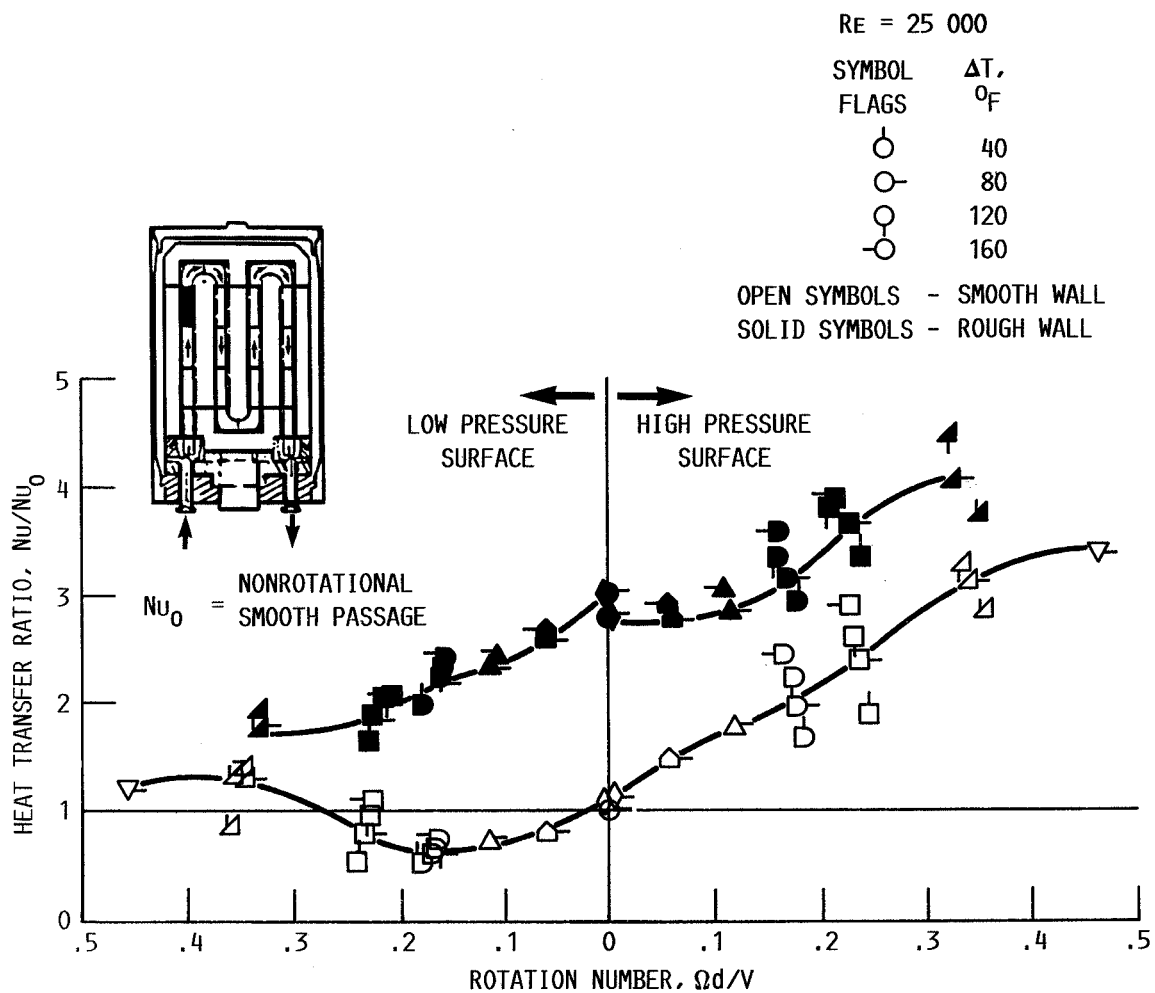
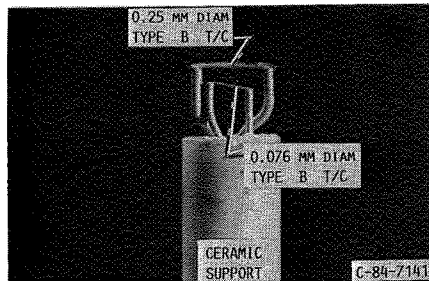
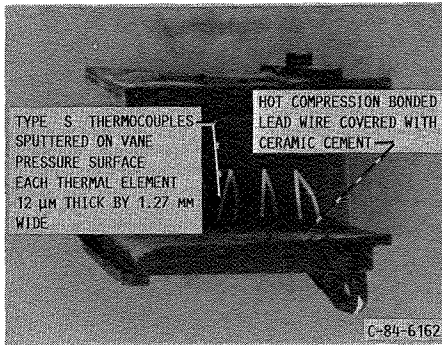


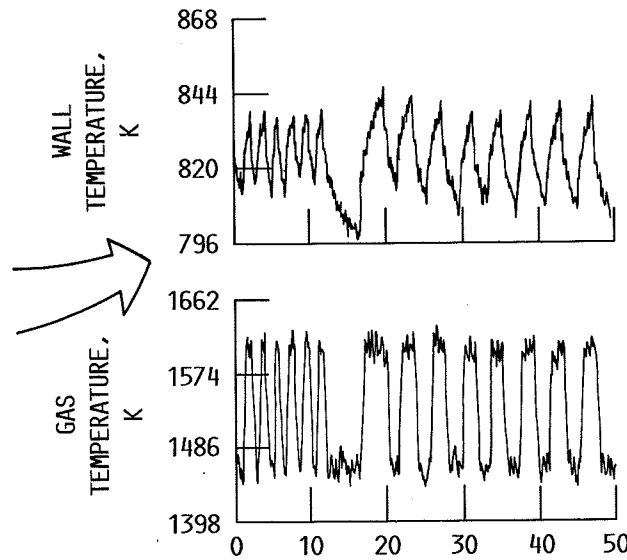
FIG. 18 THE EFFECTS OF ROTATION ON HEAT TRANSFER IN MULTIPASS COOLANT PASSAGES WITH AND WITHOUT TURBULATORS ARE SHOWN FOR THE FIRST OUTWARD FLOWING LEG.

EXPERIMENTAL TEMPERATURE MEASUREMENT



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WALL TEMPERATURE RESPONSE TO GAS TEMPERATURE RAMP



LOCAL HEAT TRANSFER
COEFFICIENTS DETERMINE FROM
DYNAMIC SIGNAL ANALYSIS

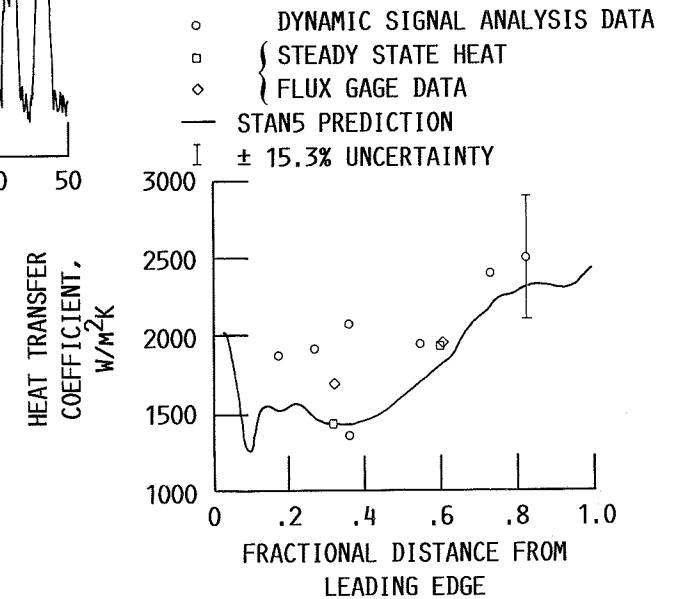


FIG. 19 HEAT FLUX MEASUREMENTS MADE IN A SIMULATED REAL ENGINE ENVIRONMENT ON STATOR AIRFOILS.

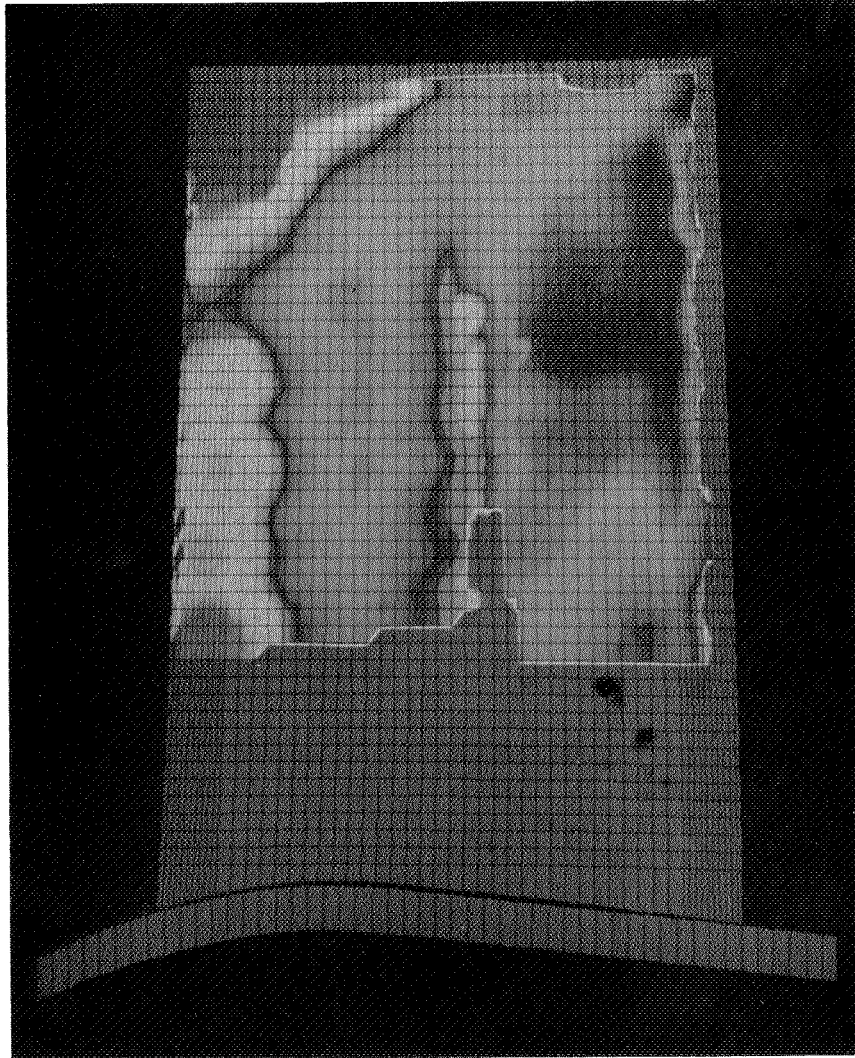


FIG. 20 TEMPERATURE DISTRIBUTION ON A ROTOR BLADE IN REAL OPERATING TURBINE ENGINE, USING A SCANNING OPTICAL PYROMETER.

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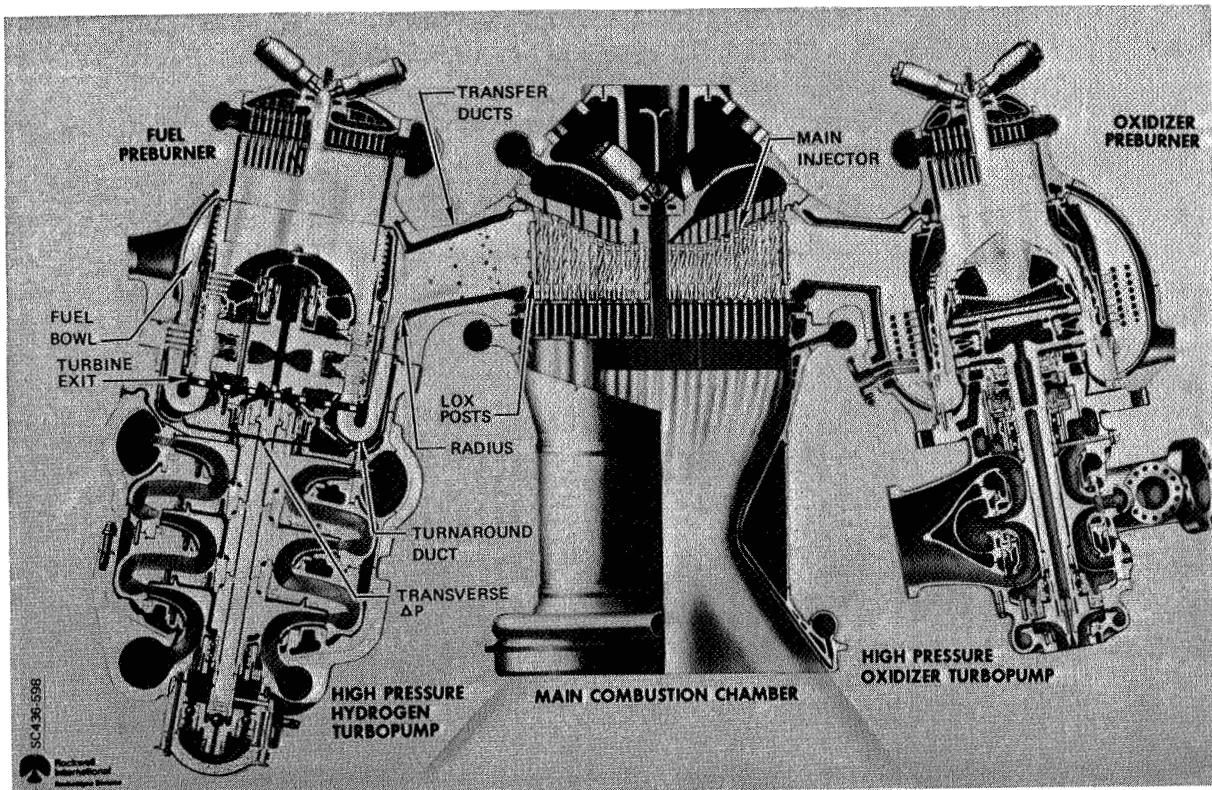


FIG. 21 SSME POWERHEAD COMPONENT ARRANGEMENT.

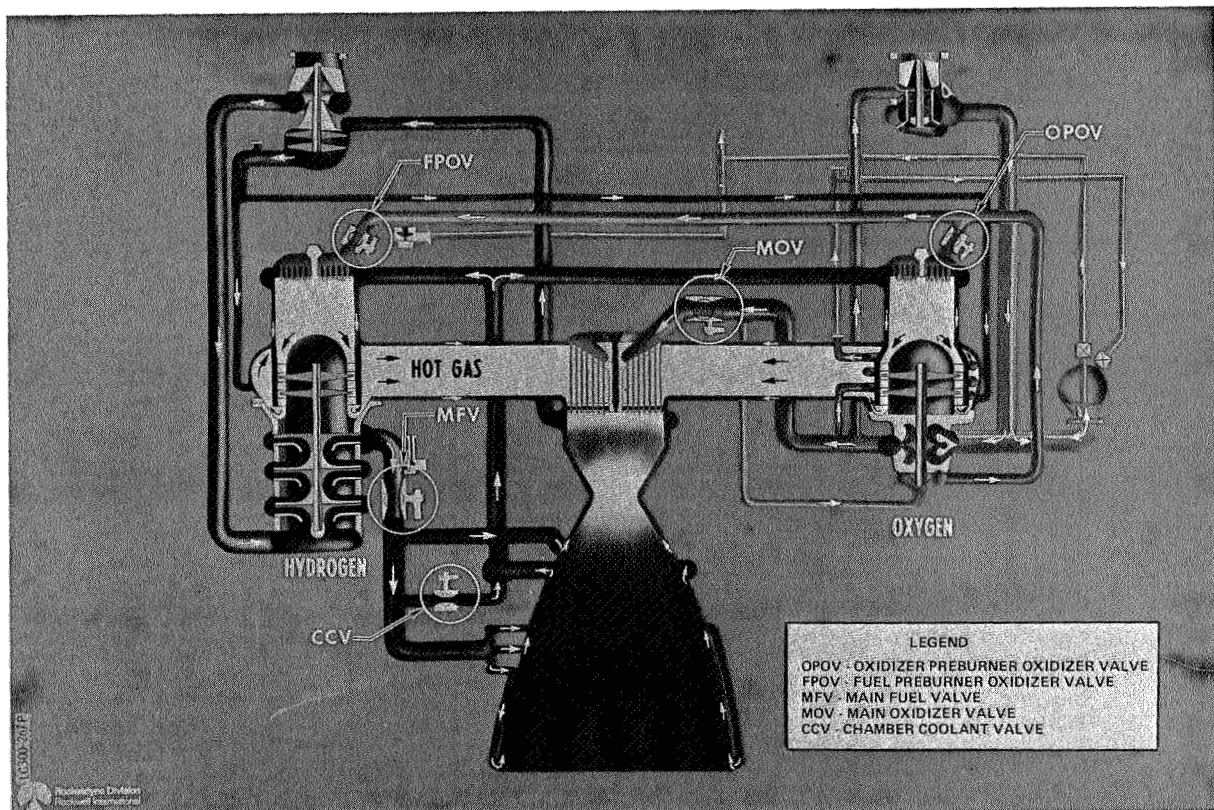


FIG. 22 SSME PROPELLANT FLOW SCHEMATIC AT FULL POWER LEVEL.

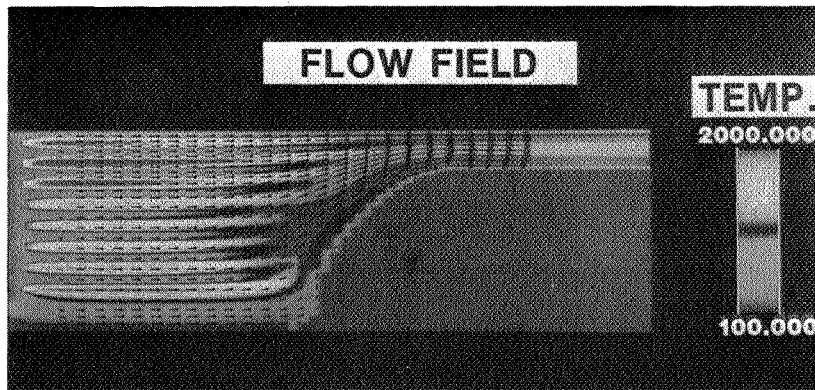


FIG. 23 CALCULATION OF THE RADIAL TEMPERATURE DISTRIBUTION FLOWING FROM THE FUEL PUMP PREBURNER INTO THE TURBINE.

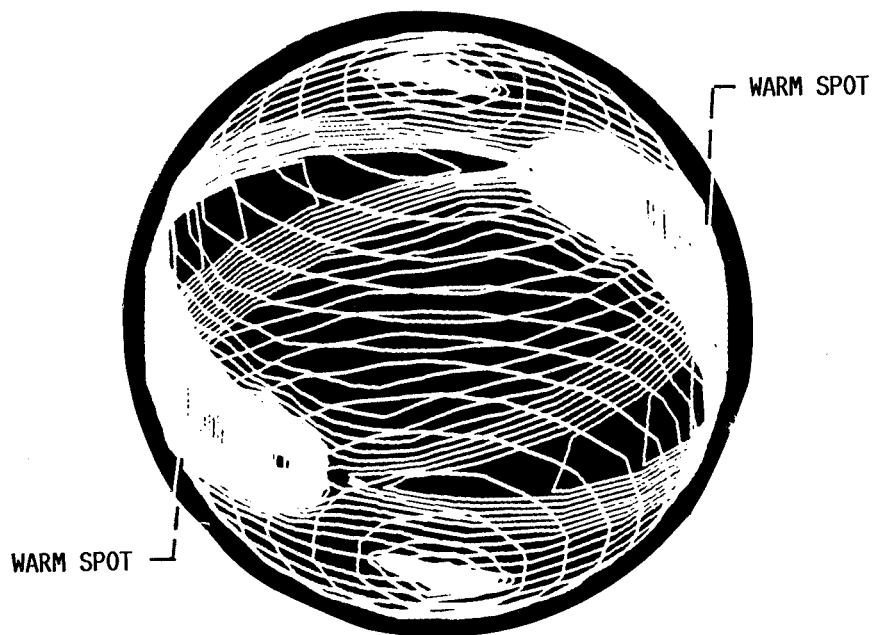


FIG. 24 PERSPECTIVE VIEW OF A CALCULATION OF THE TEMPERATURE DISTRIBUTION WITHIN A BALL BEARING ELEMENT IN THE LOX TURBO-PUMP.

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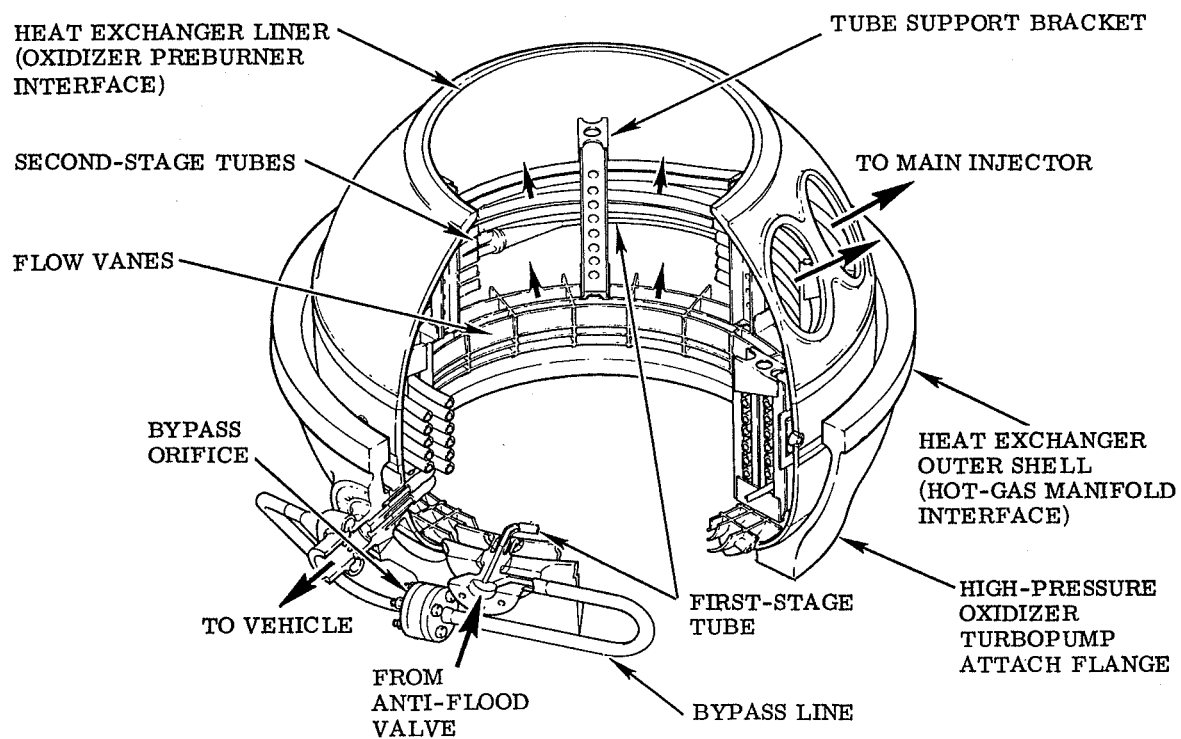


FIG. 25 LOX-60X HEAT EXCHANGER FOR SSME OXIDIZER PUMP ASSEMBLY.

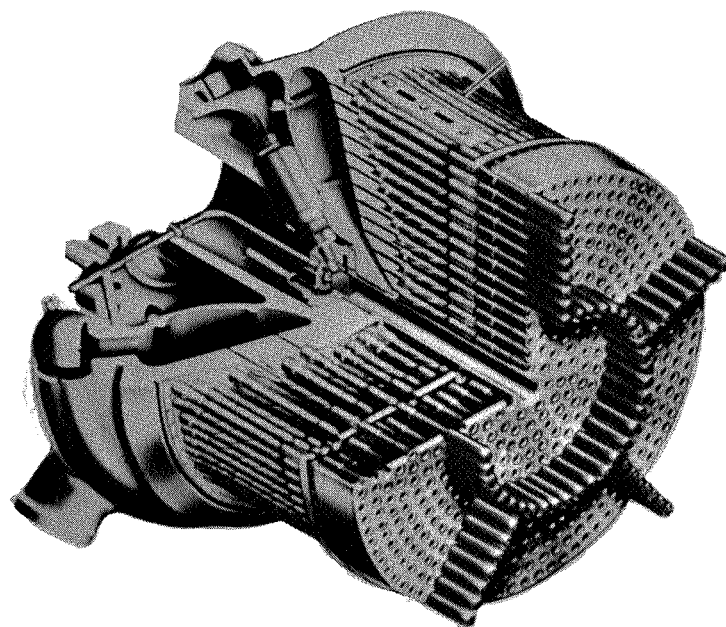


FIG. 26 MAIN COMBUSTION CHAMBER INJECTOR FOR SSME ENGINE.

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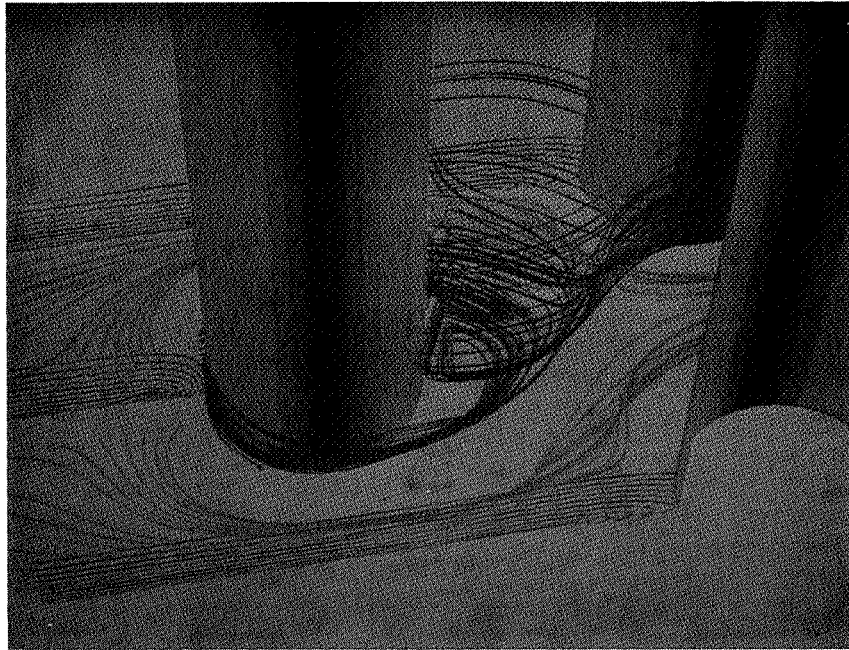


FIG. 27 PARTICLE TRACES NEAR THE POST-PLATE JUNCTION FOR TWO ROWS OF POSTS IN THE COMPUTATIONAL ANALYSIS OF THE LOX POST TYPE GEOMETRIES.

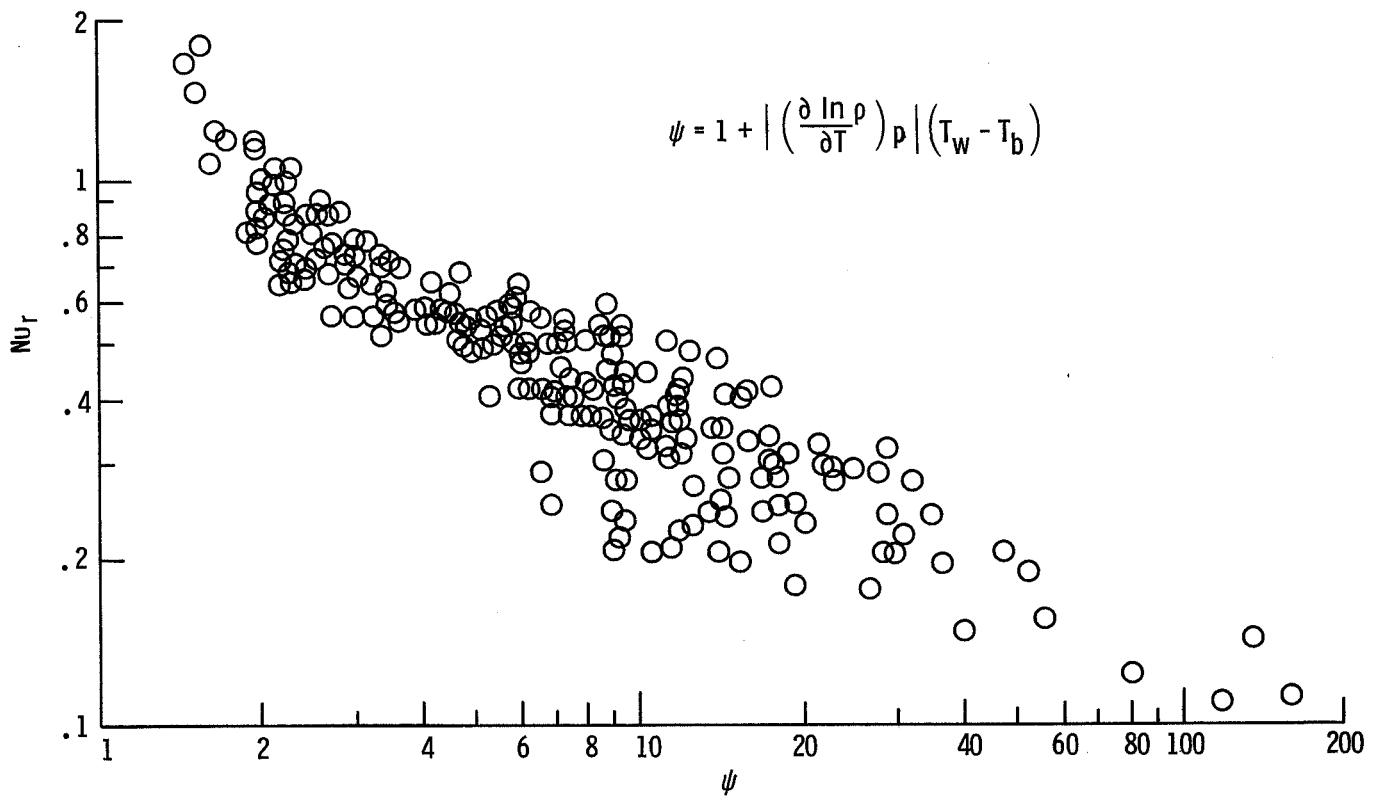


FIG. 28 CORRELATION OF HYDROGEN HEAT TRANSFER DATA IN THE NEAR-CRITICAL REGION BY ACCOUNTING FOR STRONG DENSITY VARIATIONS.

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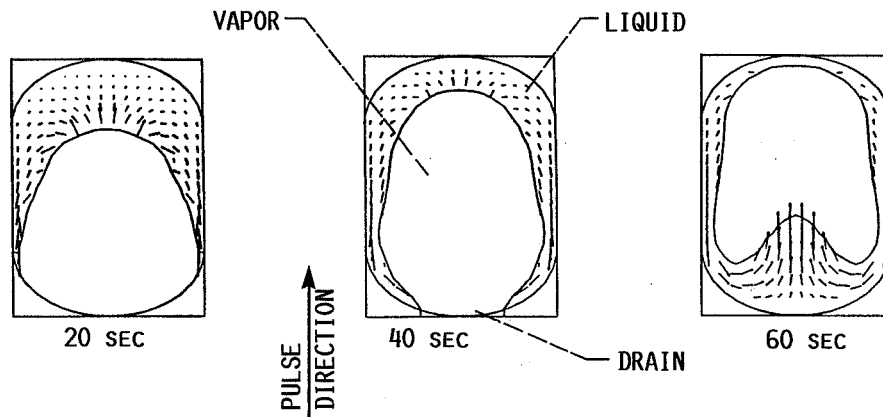
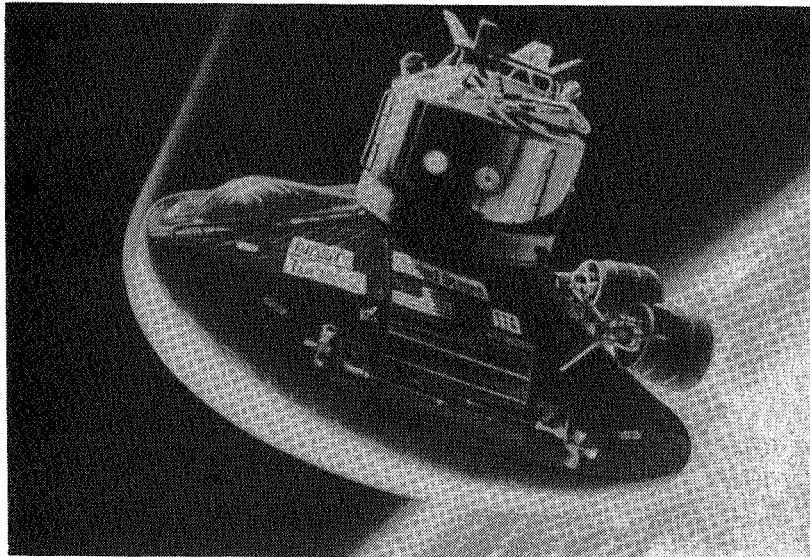
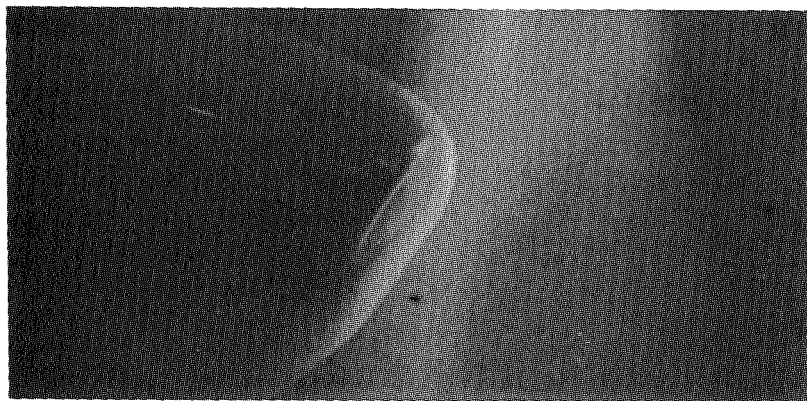


FIG. 29 COMPUTATION OF CHANGING FLUID ORIENTATION IN A PROPELLANT USING A SMALL PULSED FREQUENCY THRUSTOR (0.1 SEC DURATION AT 0.1 Hz).

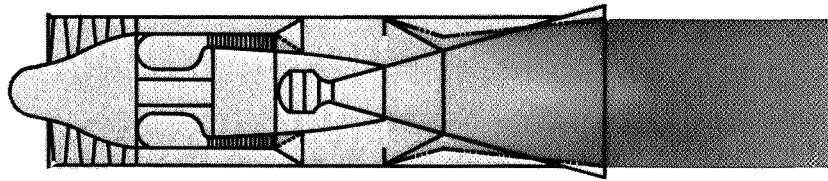


(a) ARTIST'S SKETCH.

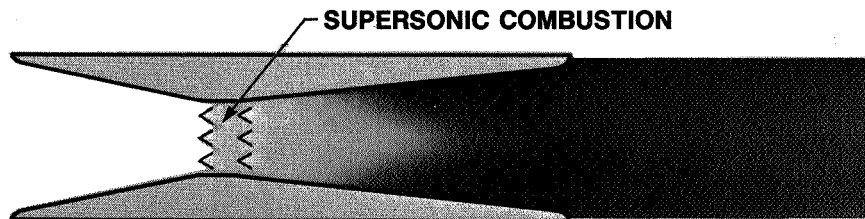


(b) EXPERIMENTAL CONFIGURATION BEING WIND-TUNNEL TESTED.

FIG. 30. AEROBRAKING OF AN ORBITAL TRANSFER VEHICLE.



(a) AIRTURBO RAMJET/ROCKET.



(b) SCRAM JET.

FIG. 31 CONCEPTUAL CROSS-SECTIONS OF POTENTIAL ENGINES FOR HYPERSONIC PROPULSION.

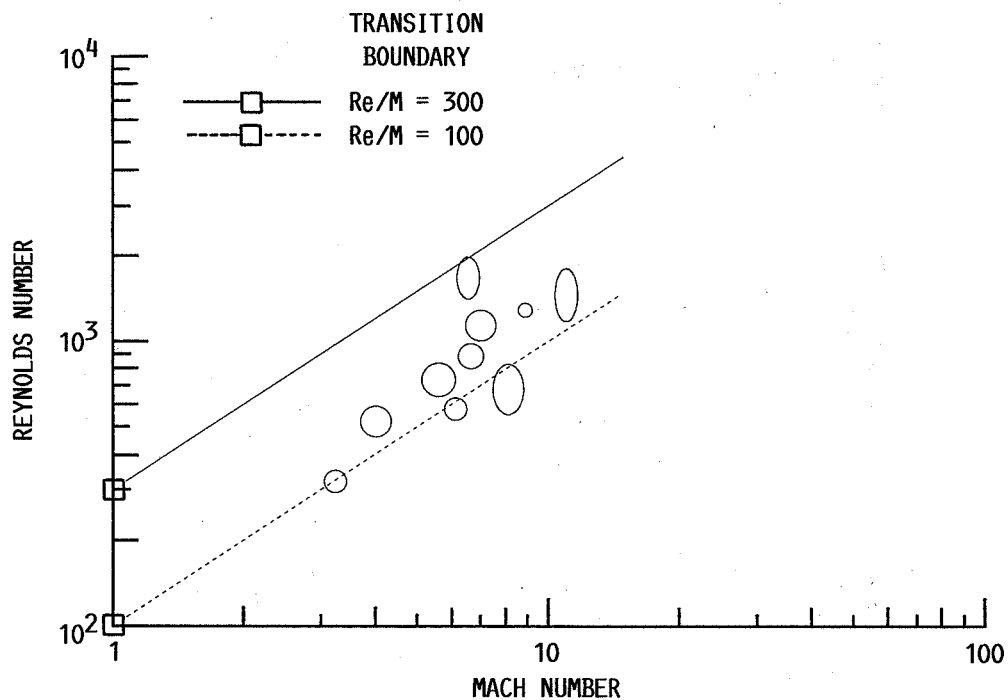


FIG. 32 BOUNDS ON BOUNDARY LAYER TRANSITION CRITERIA FOR HIGH MACH NUMBER FLIGHT.

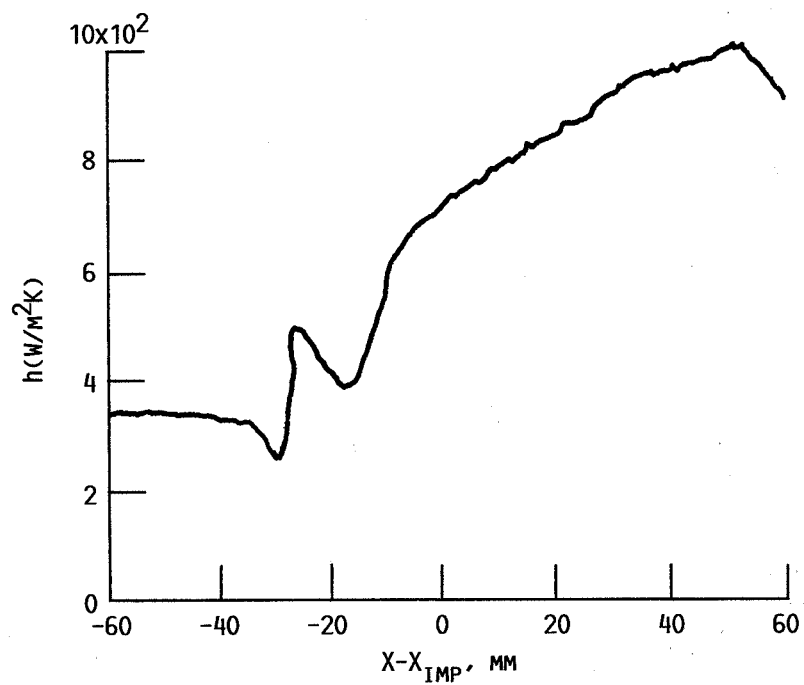
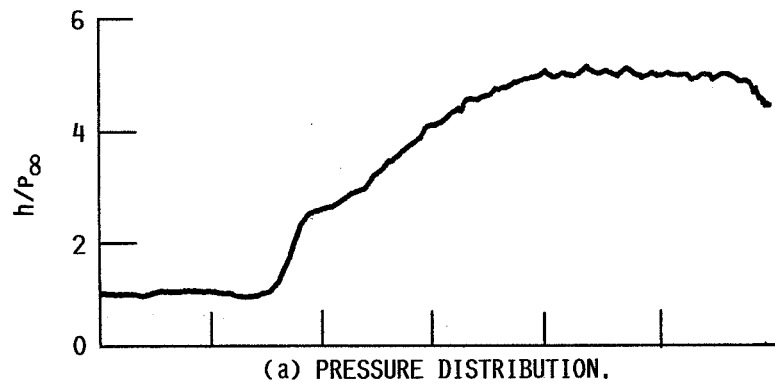


FIG. 33 EFFECT OF AN OBLIQUE (20.2°) SHOCK ON A SURFACE.

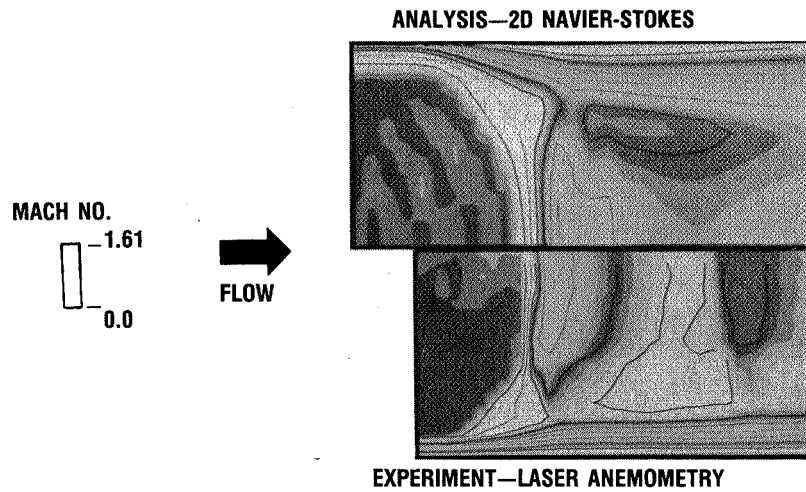


FIG. 34 COMPARISON OF ANALYSIS AND LDV MEASUREMENTS FOR A NORMAL SHOCK WAVE BOUNDARY LAYER INTERACTION.

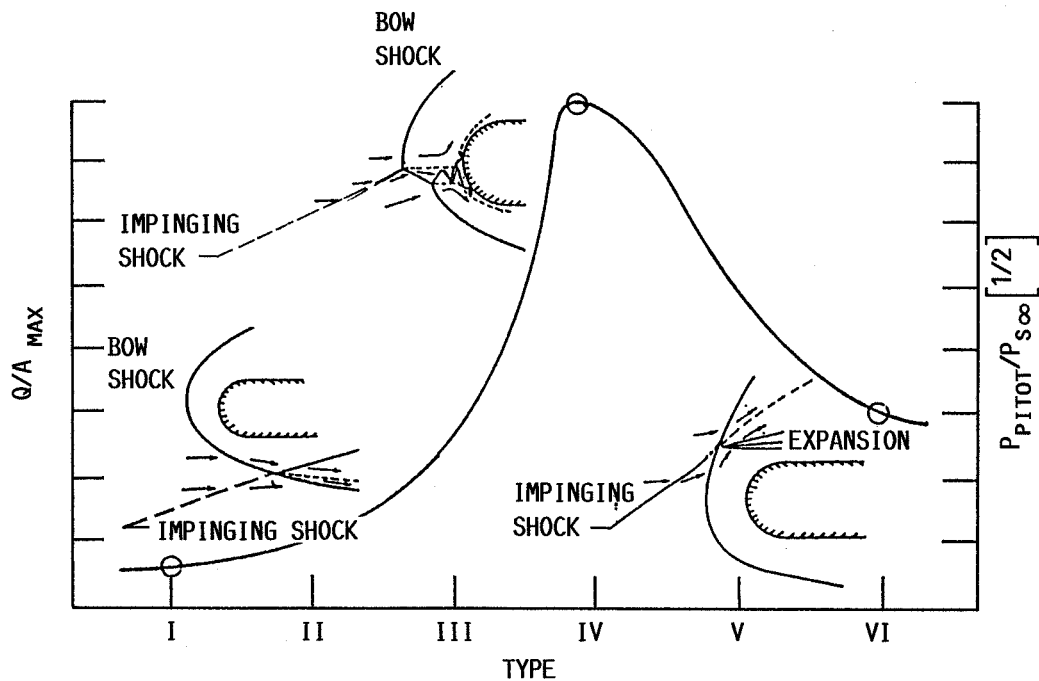


FIG. 35 REGIONS OR CLASSES OF SHOCK-ON-SHOCK INTERFERENCES AND THEIR INFLUENCE ON PRESSURE AND HEAT TRANSFER.

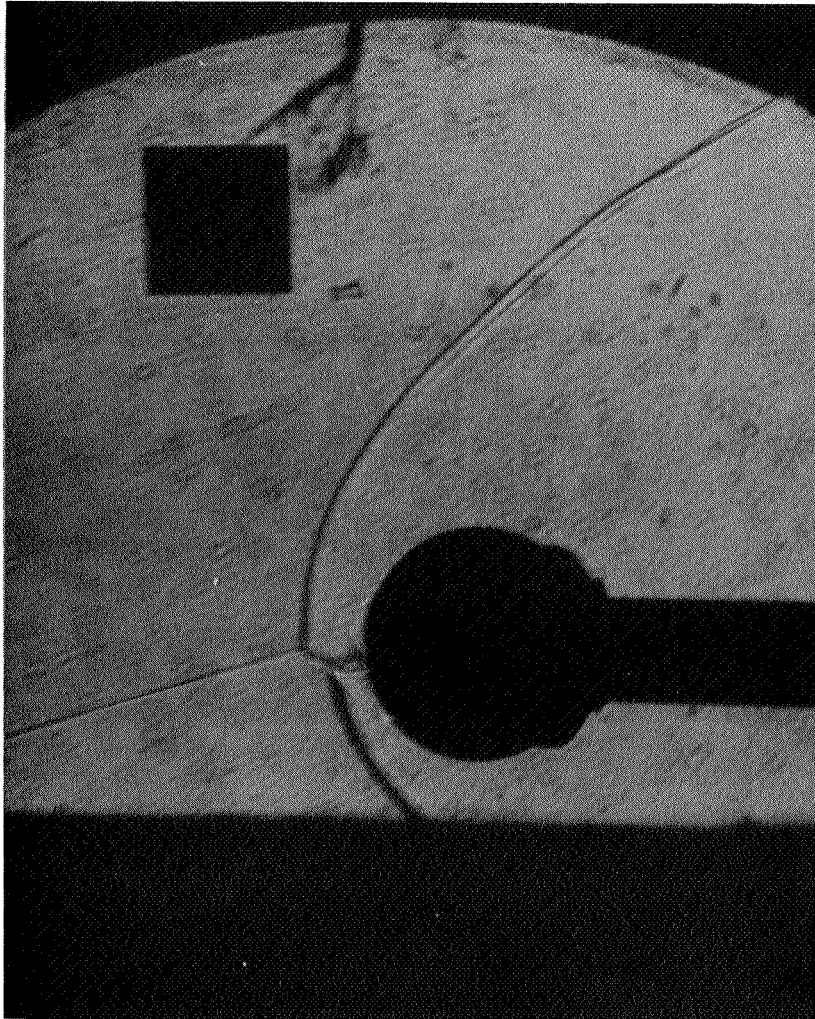


FIG. 36 SCHLIEREN PHOTOGRAPH OF A TYPE IV SHOCK-ON-SHOCK
INTERFERENCE PATTERN.

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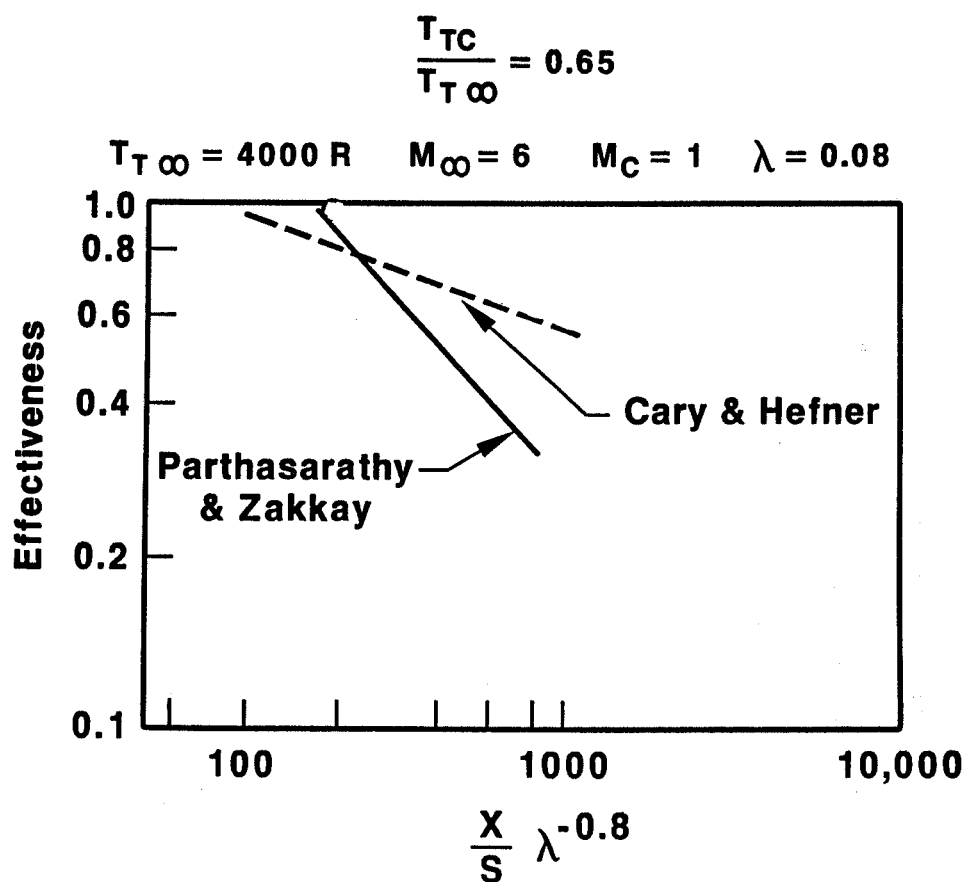


FIG. 37 RANGE OF SLOT FILM COOLING EFFECTIVENESS DATA FOR HIGH MACH NUMBER FLOW.

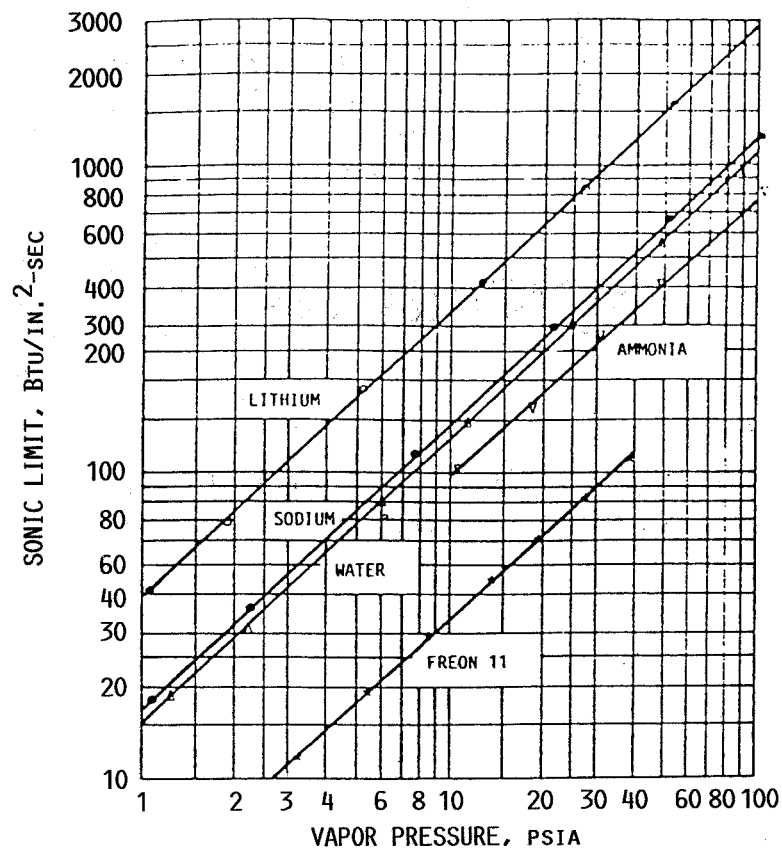
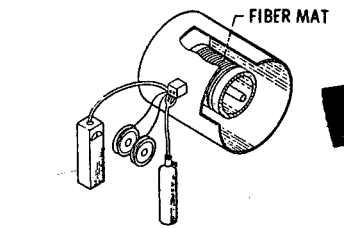
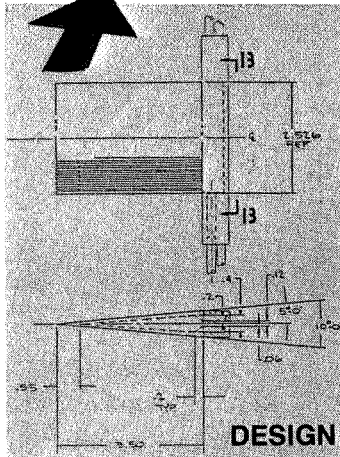


FIG. 38 THEORETICAL LIMITS FOR VARIOUS FLUIDS TO ACT AS A HIGH ENERGY HEAT REMOVAL MEDIUM IN A HEAT PIPE.

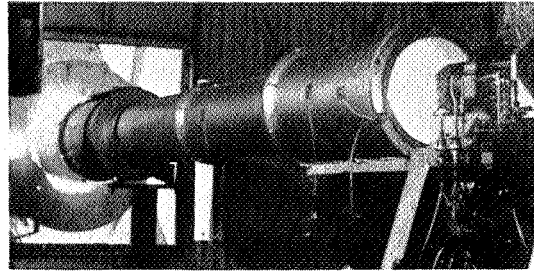
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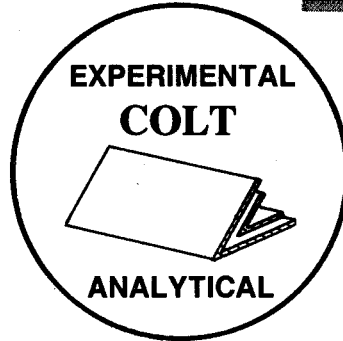
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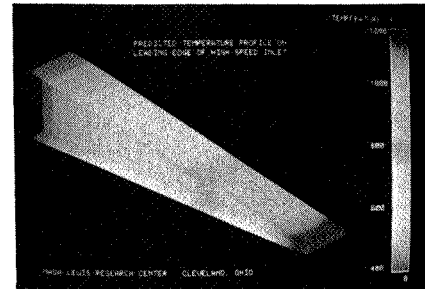


TESTING



EXPERIMENTAL
COLT

ANALYTICAL



ANALYSIS



FIG. 39 INTERDISCIPLINARY STRUCTURES, FLUIDS, MATERIALS, DESIGN, AND INSTRUMENTATION RESEARCH METHODOLOGY FOR VERY HIGH TEMPERATURE, HIGH HEAT LOAD ENVIRONMENTS.

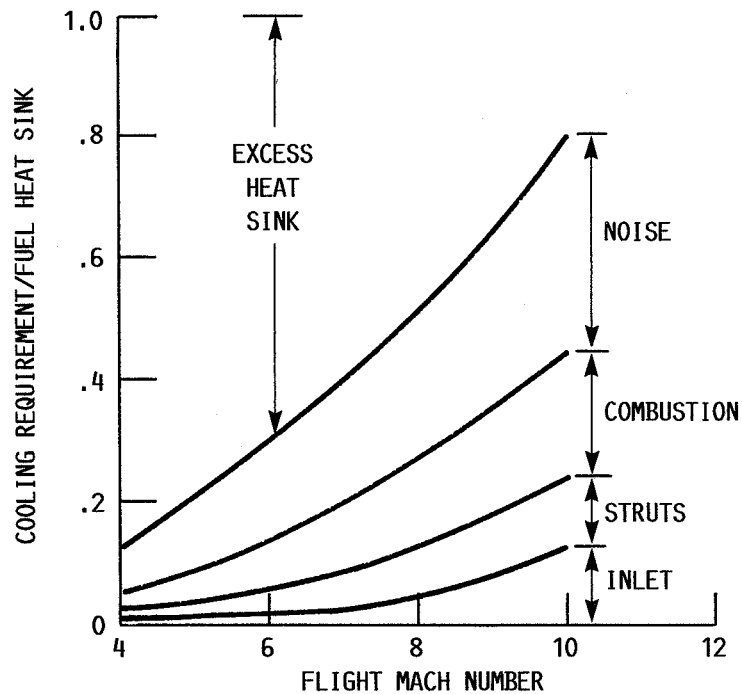


FIG. 40 PERCENT OF THE FLOWING FUEL NEEDED TO PROVIDE A HEAT SINK FOR ENGINE/VEHICLE HEAT LOADS.

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4. Title and Subtitle Heat Transfer in Aerospace Propulsion				5. Report Date	
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				10. Work Unit No. 505-62-21	
9. Performing Organization Name and Address National Aeronautics and Space Administration Lewis Research Center Cleveland, Ohio 44135-3191				11. Contract or Grant No.	
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12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D.C. 20546-0001				14. Sponsoring Agency Code	
15. Supplementary Notes Plenary lecture prepared for the National Heat Transfer Conference cosponsored by the American Society of Mechanical Engineers, the American Institute of Chemical Engineers, and the American Nuclear Society, Houston, Texas, July 24-27, 1988.					
16. Abstract This paper presents an overview of heat transfer related research in support of aerospace propulsion, particularly as seen from the perspective of the NASA Lewis Research Center. For this paper, aerospace propulsion is defined to cover the full spectrum from conventional aircraft power plants through the Aerospace Plane to space propulsion. The conventional subsonic/supersonic aircraft arena, whether commercial or military, relies on the turbine engine. A key characteristic of turbine engines is that they involve fundamentally unsteady flows which must be properly treated. Space propulsion is characterized by very demanding performance requirements which frequently push systems to their limits and demand very tailored designs. The hypersonic flight propulsion systems are subject to very severe heat loads and the engine and airframe are truly one entity. The impact of the special demands of each of these aerospace propulsion systems on heat transfer will be explored in this paper.					
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